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# RESEARCH MEMORANDUM

## ANALYSIS OF ROCKET, RAM-JET, AND TURBOJET ENGINES FOR SUPERSONIC PROPULSION OF LONG-RANGE MISSILES

## III - RAM-JET ENGINE PERFORMANCE

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# ANALYSIS OF ROCKET, RAM-JET, AND TURBOJET ENGINES FOR SUPERSONIC PROPULSION OF LONG-RANGE MISSILES

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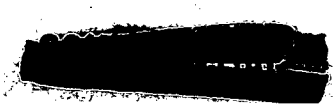
## SUMMARY

Ram-jet engine performance data are presented over a range of engine design variables to permit selection and evaluation of a ram-jet engine configuration for a long-range supersonic missile. Calculated design-point performance of engines using JP4 fuel is presented for a wide range of engine total-temperature ratios and combustion-chamber-inlet Mach numbers for flight Mach numbers from 1.5 to 4.0. The results include engine thrust, drag, fuel consumption, area ratios, and weight, and are suitable for use in design studies of missiles incorporating either internally or externally mounted ram-jet engines. Maximum engine fuel specific impulse (including nacelle drag) is approximately 1600 pound-seconds per pound and occurs at a flight Mach number of 2.5. Over-all engine efficiency, however, continues to increase to a flight Mach number of 4.0, where it is 35 percent.

Important gains in both thrust coefficient and specific impulse may be achieved by improving the diffuser pressure recovery. Changes in flame-holder pressure loss and combustor length have only small effects on engine performance. That these factors, however, influence combustion efficiency is significant, because the specific impulse varies directly with the efficiency, although the thrust coefficient is practically unaffected. Engine performance is very sensitive to changes in nozzle performance, a 1-percent variation in velocity coefficient often producing a 3-percent variation in engine thrust and specific impulse. Some underexpansion of the exhaust gases is desirable to reduce nacelle drag whenever the nozzle-exit diameter exceeds that of the combustion chamber.

With a fixed-geometry configuration, a ram-jet engine does not operate satisfactorily at off-design conditions. Somewhat better thrust can be obtained with the added complication of a translating-spike diffuser, although the specific impulse is poorer. Use of a movable plug to vary the throat area of the exhaust nozzle yields both thrust and specific impulse approaching that of a continuously variable-geometry engine.



# INTRODUCTION

## The Complete Study

The role of guided missiles in the nation's weapons system has received much attention in recent years. Latest advances in research and development of engines, aerodynamics, and guidance make practicable the utilization of these missiles for delivery of a warhead at supersonic speeds to a target thousands of miles distant. As an aid to the solution of development and design problems, it is the purpose of this series of reports to study the potentialities of various engines suitable for the supersonic propulsion of long-range missiles and to determine those characteristics which result in the best over-all performance of the engine and missile combination. In order to keep primary emphasis on the characteristics of the engine proper, the material on engine performance is separated from that which considers the over-all missile system. The rocket-engine performance and the study of the rocket-powered missile are presented in references 1 and 2, respectively. The performance of the ram-jet engine is presented herein.

Continuing research indicates many improvements that are possible in some of the engine components. The performance of the components selected for the engines in this analysis has either been demonstrated in the laboratory or appears, from available data, to be certain of attainment within a reasonable time. Similarly, advanced features of airframe design that are believed possible to develop in a comparable time were also selected.

The principal mission to which attention has been directed is that of a long-range strategic bombardment missile. The configurations studied are all limited to simple two-stage designs consisting of a rocket booster used only during the initial phase of flight and a second stage that flies the remaining distance under its own power. Missiles propelled by the air-breathing engines are considered for cruise flight Mach numbers from 2 to 4 at altitudes from 35,000 to 80,000 feet. The rocket-propelled missiles are considered to travel along a ballistic trajectory, even though there are serious problems of re-entry into the atmosphere. Although glide and even skip rockets have frequently been proposed for this application, the many problems and uncertainties associated with the aerodynamics of these air-borne types preclude them from the present study.

Rocket engines for long-range ballistic missiles have very short operating times (1 to 3 min) and during this time are operating at continually differing flight conditions. The study of the rocket engine (ref. 1) must, therefore, be based on a compromise design. Fortunately, however, the study is simplified, because the performance of a rocket engine is practically independent of flight velocity. On the other hand, air-breathing engines for long-range missile propulsion generally operate continuously at their design conditions for the major portion of the flight time. Therefore, even though off-design engine performance is considered in these analyses, the study of the air-breathing engines can be based principally on design-point performance.

## Ram-Jet Engine Performance

The purpose of this report is to present sufficient ram-jet engine performance data over a range of engine design parameters to permit selection and evaluation of a ram-jet engine configuration for a long-range supersonic missile. A second purpose is to illustrate the relative importance of the different engine parameters that influence the design of the ram-jet engine. Many other thermodynamic cycle studies of ram-jet engines have been presented in the literature (e.g., refs. 3 and 4). This report presents a wider range of operating conditions, uses somewhat advanced, but realistic, component characteristics, and demonstrates the effect of changes in these component characteristics.

The missiles are considered to have externally mounted engines. Therefore, nacelle drag is included in the engine performance data to facilitate separate consideration of airframe and engine. However, engine drag coefficients are tabulated so that the results may also be used without drag for other studies.

The report has three major sections:

(1) General design-point engine performance is presented for a wide range of engine total-temperature ratios and combustion-chamber-inlet Mach numbers for flight Mach numbers from 1.5 to 4.0. The fuel used is JP4, and nominal assumptions are used for the component characteristics.

(2) The sensitivity of these design-point results to changes in the nominal assumptions is indicated by showing the effect of varying the different component parameters, one at a time, on the performance of selected engines. The parameters investigated are diffuser pressure recovery, flame-holder pressure loss, combustion efficiency, combustor length, nozzle velocity coefficient, nozzle expansion ratio, nozzle jet-deflection angle, and altitude. Calculations were also made comparing a high-energy fuel (pentaborane) with JP4.

(3) Off-design performance is presented for an engine designed for efficient cruising at a flight Mach number of 3.5. Configurations having fixed geometry, continuously variable geometry, and two types of practically variable geometry are compared. Off-design performance is not emphasized, because results obtained in missile studies indicate that self-boosting capabilities are not important for ram-jet-powered long-range missiles.

## ANALYSIS

The symbols used in this report are defined in appendix A.

A schematic diagram of the ram-jet engine is shown in figure 1. High-speed air enters the engine at station 1 and is decelerated to a low velocity at 2. Fuel is added, ignition takes place, and combustion is stabilized at the flame holders, between stations 2 and 3. Combustion occurs in a constant-area duct from stations 3 to 4, and the hot gases are expanded and discharged through a convergent-divergent nozzle (stations 4 to 6).

The performance calculations were made on the basis of one-dimensional flow, using the equations of state, continuity, and conservation of momentum and energy. Reference 4 presents equations similar to those used in the present analysis, in which the values of $\gamma$ are based on the gas static temperature and composition at each station. The problem of specifying the gas properties is discussed more fully in appendix B, and the assumed engine geometry and methods used in calculating the engine drags are detailed in appendix D. Engine performance data, which are generally presented for an altitude of 70,000 feet, may be used for any altitude in the isothermal region of the atmosphere with negligible error.

Engine performance is presented in this report in terms of the following:

(1) Propulsive thrust coefficient $C_T$, defined as engine thrust minus nacelle drag per unit cross-sectional area, divided by free-stream incompressible dynamic pressure. The cross-sectional area used is the diffuser capture area or the combustion-chamber frontal area, whichever is larger. This coefficient is a measure of the engine size required to produce a given amount of propulsive thrust.

(2) Specific impulse I, defined as engine thrust minus nacelle drag divided by engine fuel-flow rate. At any given flight speed, this parameter is a measure of the efficiency with which thrust is produced.

Appendix C describes a simple method of estimating engine weight, from which it was concluded that ram-jet engine weight per unit cross-sectional area is approximately constant.

Most of the various engine parameters fall into two groups: (A) those of major importance that cannot be finally specified without a complete missile flight analysis, and (B) those that can be realistically chosen from an engine study alone, or that are limited by what can practically be achieved. These major variables (group A) were taken as flight Mach number, combustion-chamber-inlet Mach number, and engine total-temperature ratio. A complete set of design-point performance calculations was obtained for different values of these variables, based on nominal assumptions made for the component parameters of group B. The range of calculations for these major variables included flight Mach numbers from 1.5 to 4.0, combustion-chamber-inlet Mach numbers from 0.125 to 0.225, and engine total-temperature ratios corresponding to fuel-air ratios of approximately 0.01 to stoichiometric, except where limited by thermal choking.

The following values were used for the variables of group B in the design-point calculations:

(1) Diffuser pressure recoveries were assumed for a single-cone spike diffuser with a kinetic-energy efficiency of 90 percent in the subsonic portion. Cone angle was varied with design flight speed to achieve maximum pressure recovery. The design-point engines operated critically with no spillage. Reference 5 shows that, in the present state of inlet development, the engine performance obtained with this single-cone inlet with low-drag cowl is as good as that afforded by more elaborate inlets such as the isentropic spike. Figure 2 shows the assumed variation of pressure recovery with flight Mach number for critical operation. These values are in good agreement with the experimental data for similar inlets reported in reference 6.

(2) Flame-holder total-pressure loss was taken as twice the incompressible dynamic pressure at station 2. Combustion of the fuel (JP4 with a lower heating value $H$ of 18,640 Btu/lb) took place from stations 3 to 4 with an assumed efficiency of 0.90. The resulting relation between $\tau$ (engine total-temperature ratio) and fuel-air ratio $f/a$ for different flight Mach numbers is shown in figure 3. Reference 7 reports the achievement of about 0.95 efficiency with the same amount of flame-holder loss in tests of a 16-inch combustor at a combustor pressure of about 1 atmosphere. The combustor pressure, however, in an engine for long-range missiles may for some flight paths be subatmospheric.

(3) The nozzle velocity coefficient was taken as 0.975. Values of this magnitude were obtained experimentally in large-scale tests of convergent-divergent nozzles at nozzle pressure ratios $P_4/p_6$ of about

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5 (ref. 8). Recent unpublished results indicate that this value may be approached even at high nozzle pressure ratios. At flight Mach numbers of 2.5 and higher, the nozzle-exit diameter is generally the largest diameter of the engine. To reduce nacelle drag in these cases, the nozzle expansion ratio was made less than that required for complete expansion of the gases to ambient pressure. From data obtained in missile studies showing the effect of expansion ratio on missile range, the optimum expansion ratio is roughly generalized by the empirical formula

$$\frac{A_6}{A_3} = 1 + 0.55\left[\left(\frac{A_6}{A_3}\right)_{p_6=p_0} - 1\right] \tag{1}$$

where $\left(\frac{A_6}{A_3}\right)_{p_6=p_0}$ is the ratio of nozzle-exit to combustion-chamber area for complete expansion. This expression was used for the design-point engine calculations whenever the nozzle-exit diameter exceeded that of the combustion chamber. In all other cases the nozzle was made completely expanding.

The sensitivity of the design-point results to changes in these assumed values of the various component parameters was indicated by calculating the effect of varying these parameters, one at a time, at flight Mach numbers of 2.5 and 3.5. At each speed, two values of $\tau$ were considered, a low value for good cruising performance and a higher value to give increased thrust for missile acceleration.

The off-design performance of engines designed for cruising at a flight Mach number of 3.5 was also evaluated. Engines equipped with the following features were considered:

(1) Continuously variable diffuser and nozzle

(2) Variable-throat-area nozzle with fixed diffuser

(3) Translating-spike diffuser with fixed nozzle

(4) Fixed diffuser and nozzle

## RESULTS AND DISCUSSION

### Design-Point Performance

The calculated design-point values of propulsive thrust coefficient and specific impulse are shown in figure 4 as functions of flight Mach number $M_0$, ratio of combustion-chamber-exit to -inlet total temperature $\tau$, and combustion-chamber-inlet Mach number $M_2$. These data, as well as

engine area ratios and drag coefficients, are listed in table I. The values of $C_T$ and I include nacelle drag. The corresponding values without nacelle drag may be obtained by the relations

$$C_F = C_T + C_D \tag{2}$$

$$I_F = I \frac{C_F}{C_T} \tag{3}$$

Performance of engines having velocity coefficients other than 0.975 may be calculated from the following formula:

$$C_T = \frac{C_V}{0.975}\left[(C_T)_{0.975} + C_D + 2\left(\frac{A_1}{A_3}\right)\right] - C_D - 2\left(\frac{A_1}{A_3}\right) \tag{4}$$

which is based on the assumption that the jet thrust is directly proportional to the velocity coefficient and that the nacelle drag does not change.

Performance can also be computed for values of combustion efficiency other than 0.90. At any constant value of $\tau$, the thrust coefficient remains essentially constant with changes in combustion efficiency, and specific impulse and fuel-air ratio are given by

$$I = \frac{\eta_c}{0.90} I_{0.90} \tag{5}$$

$$\frac{f}{a} = \frac{\eta_c}{0.90}\left(\frac{f}{a}\right)_{0.90} \tag{6}$$

Figure 4 shows that high thrusts are obtained at the high values of $\tau$ and maximum specific impulses at intermediate values of $\tau$. Raising $\tau$ (at a constant $M_0$ and $M_2$) increases the exit momentum of the gases, mainly because of the higher jet velocity and, to a lesser degree, because of the increased fuel mass flow. However, as $\tau$ is raised, the fuel flow increases at a greater rate than the jet thrust; so that, after the constant loss of the inlet momentum drag is sufficiently overcome, the specific impulse reaches a maximum and then decreases. When nacelle drag is included, the value of $\tau$ for maximum specific impulse is raised.

The effect of flight Mach number on over-all engine efficiency, propulsive thrust coefficient, and specific impulse is shown in figure 5, in which the value of $\tau$ is varied to provide maximum I and E at each flight speed. Combustion-chamber-inlet Mach number is generally 0.200, except for $M_0$ above 3.35, where it was necessary to reduce $M_2$ to prevent the diffuser-inlet diameter from exceeding that of the combustion

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chamber. Maximum I (1600 lb-sec/lb including nacelle drag) occurs at $M_0$ of 2.5. This flight Mach number, however, may not be optimum for a missile, because missile range is more nearly related to the over-all engine efficiency, which in turn is proportional to the product of specific impulse and flight velocity rather than to specific impulse alone. Maximum E of the order of 0.35 is realized at $M_0$ near 4, while the highest efficiency obtainable at $M_0$ of 2.5 is only 27 percent.

The effect of combustion-chamber-inlet Mach number $M_2$ is indicated in figure 4 but is more readily apparent in a cross plot of some of these data (fig. 6). The propulsive thrust coefficient increases with $M_2$ because of the essentially linear increase in air flow to the point where the inlet capture area becomes equal to the combustion-chamber area. The value of $M_2$ at which these areas are equal is higher than the region of interest shown in figure 6, but is a function of the flight speed and the diffuser pressure recovery. Raising $M_2$ also increases both the flameholder pressure loss and the momentum pressure loss due to heat addition. If nacelle drag were not included, the specific impulse would decrease with increasing values of $M_2$. However, higher values of $M_2$ reduce the diameter of the combustor and nozzle relative to the diffuser capture area and so result in lower nacelle drag per pound of air (provided the capture area remains smaller than the combustion-chamber area). Because of these two opposing effects, the specific impulse is fairly insensitive to variations in $M_2$ for the conditions of figure 6.

## Effect of Variations in Design-Point Assumptions

Diffuser pressure recovery. - Diffuser total-pressure ratio is used in this report as a measure of the efficiency with which the diffuser converts the kinetic energy of the captured air stream to pressure. Lines of constant kinetic-energy efficiency superimposed on the curve of pressure recovery against flight Mach number (fig. 2) show that the lower numerical values of total-pressure ratio at high flight Mach numbers do not necessarily mean lower diffuser efficiency.

The nominal diffuser assumed for the design-point calculations is an oblique-shock inlet with a single-cone spike centerbody. Figure 7 shows the effect on engine performance of changes in the assumed values of pressure recovery. (This performance is based on the drag of a low-angle cowl at all pressure recoveries.) The engine air flow per unit combustion-chamber area increases linearly with pressure recovery, so that the propulsive thrust coefficient (based on combustion-chamber area) also increases nearly linearly. As pressure recovery is increased, the diffuser capture area enlarges relative to the combustion chamber in order to handle these larger air flows at constant $M_2$. At high flight Mach numbers and high pressure recoveries, the resulting capture area often becomes greater than the combustion-chamber area. The size of the engine

required to produce a given thrust is then indicated by basing the thrust coefficient on diffuser capture area. At a flight Mach number of 2.5, the combustion chamber is always the larger for the range of pressure recoveries considered. At a flight Mach number of 3.5 and $M_2$ of 0.20, the capture area becomes the larger at pressure recoveries greater than 0.43, which causes the sharp break in thrust coefficient observed at this point in figure 7. The engine specific impulse increases with increasing pressure recovery because of the higher pressure ratio across the exhaust nozzle. The higher air flow also improves the specific impulse because of the lower nacelle drag per pound of air.

In general, however, diffuser designs that result in improved pressure recoveries have associated with them high engine-cowl pressure drags. Consequently, the gain in engine performance resulting from improved pressure recovery may be largely offset by the resulting engine drag increase. Figure 8 shows engine performance at a flight Mach number of 3.5 as a function of both pressure recovery and engine nacelle drag coefficient. The dotted line repeats the low-angle-cowl drag values from figure 7 and represents the best performance attainable at any value of pressure recovery. Figure 8 indicates the penalties in drag rise that are acceptable to obtain better engine performance as a result of improved pressure recovery. Calculations based on the experimentally measured pressure recovery and cowl drags reported in reference 5 confirm the conclusion that currently available high-recovery inlets do not yield better over-all engine performance than does the single-cone type assumed throughout this analysis. Although the single-cone inlet was used to give performance representative of that available with other current inlet types, the advanced inlets have greater potentialities for improvement, as indicated in figure 8. Other factors must also be considered, of course, in comparing different diffuser designs. For example, a single-cone inlet may be easier to design and manufacture and is less sensitive to angle of attack than are more elaborate types. On the other hand, the higher pressure provided by an advanced inlet may increase combustion efficiency and prevent blow-out.

Combustion efficiency. - If $\tau$ is held constant in an engine, variations in combustion efficiency have only a negligible effect on engine thrust. However, fuel flow and hence specific impulse are directly proportional to the combustion efficiency. The great importance of this effect lies in the fact that the range of a ram-jet missile varies directly with the specific impulse, if all other factors do not change.

Flame-holder pressure loss. - The purpose of the flame holder is to ensure the ignition and efficient burning of a fuel-air mixture moving at several hundred feet per second when the laminar flame speed of the mixture may be in the order of only 5 feet per second. Increased flow blockage and turbulence often improve the combustion efficiency but introduce pressure losses detrimental to the thrust output of the engine. A compromise is often necessary between these opposing factors. The change in propulsive thrust and specific impulse with the flame-holder cold-flow

pressure-drop coefficient is indicated in figure 9 for a constant combustion efficiency.

Combustion-chamber length. - Improved combustion efficiency may also result from lengthening the combustion chamber to obtain more complete burning; however, this results in a heavier engine and increased drag. The effect of increased drag on the propulsive thrust coefficient and specific impulse due to lengthening the engine is shown in figure 10 at a flight Mach number of 3.5 (for constant combustion efficiency). The corresponding change in engine weight is also indicated. As described in appendix D, the design-point engine calculations assumed a nominal combustor length equal to 3 diameters of the combustion chamber, with a total engine length of 9 diameters. Doubling the combustor length from 3 to 6 diameters (at constant $\tau$) reduces $C_T$ and I by less than 2 percent but, according to the weight estimates of appendix C, increases the installed engine weight by 24 percent.

Fuel type. - Another combustor variable that may be changed to improve performance is the fuel used. High-energy fuels permit raising both thrust and specific impulse, but they are generally more expensive than hydrocarbon fuels, or they may have other undesirable characteristics such as pumping or storage problems. Calculations were made for pentaborane ($B_5H_9$) as a typical high-energy fuel frequently mentioned for ramjet applications. Figure 11 shows the propulsive thrust coefficient and specific impulse of an engine designed for flight Mach number of 3.5 for pentaborane and JP4. These calculations for $B_5H_9$ were made with the assumption of equilibrium composition of the exhaust gases and with expansion to the same area assumed with JP4. Data for these calculations were taken from reference 9. These curves are for a combustion efficiency of 0.95 for the pentaborane and 0.90 for the JP4 fuel.

For the same thrust coefficient, a specific impulse with pentaborane of more than 150 percent of that with JP4 is indicated at low fuel-air ratios up to those that give maximum specific impulse. The improvement is less at high fuel-air ratios that give near maximum thrust coefficient. Because of other factors, however, such as different fuel densities, missile range is not necessarily improved in the same proportion.

Nozzle area ratio. - The effect of nozzle area ratio is presented in figure 12. Very little loss in propulsive thrust coefficient and specific impulse is suffered by cutting back the nozzle area as much as 30 percent from that required for complete expansion. In fact, for smaller amounts of underexpansion, gains of 1 or 2 percent may be realized, because reducing these areas reduces the external nacelle drag sufficiently to compensate for the lower internal thrust. In addition, since the nozzle was assumed to have a velocity coefficient less than 1.0, a small amount of underexpansion results in a very small increase in internal thrust. It is sometimes proposed that the nozzle-exit area not be permitted to exceed the combustion-chamber area. This condition ($A_6 = A_3$) is marked on the curves of figure 12. It is apparent that

this amount of underexpansion results in appreciable losses, particularly at the higher flight speed. The performance of a convergent nozzle ($A_6 = A_5$) is seen to be very poor at both speeds.

Nozzle velocity coefficient. - The velocity coefficient (defined as the actual velocity at the nozzle exit divided by the ideal isentropic velocity at the nozzle exit for the same pressure ratio) is used to indicate the amount of the nozzle internal flow losses. These losses, which reduce the total pressure, are due to shocks, turbulence, and viscous effects within the gas stream and to wall friction at the gas boundaries. The effect on engine performance of a variation in the nozzle velocity coefficient from the assumed value of 0.975 is indicated in figure 13. A 1-percent change in velocity coefficient changes the thrust and specific impulse from 2 to 3 percent for the indicated values of Mach number and $\tau$.

There are other sources of thrust losses through the nozzle that do not affect total-pressure loss. In addition to the flow losses treated through the application of a nozzle velocity coefficient, the assumption of one-dimensional flow implies that all exhaust gases are discharged axially and that there are no gradients in radial velocity. Neither of these implications is necessarily true. A nonuniform temperature distribution at the combustor exit would result in radial velocity gradients even with a nozzle designed for axial discharge. Calculations indicate that all reasonable temperature distributions, such as a parabolic profile, result in thrust losses of less than 2 percent. Losses due to nonaxial discharge from a conical nozzle with a half-angle of $15^\circ$ would be of the order of 1.5 percent. Use of a smaller angle or changes in nozzle contour would reduce this loss, although possibly at the expense of increased manufacturing cost and nozzle length.

Nozzle jet-deflection angle. - An interesting possibility for improving the performance of a ram-jet missile lies in turning the jet thrust of the engine downward. This slightly decreases the forward thrust and specific impulse but provides some lift, thereby permitting the use of a smaller wing and lowering the missile weight and drag. Figure 14 presents the effect of jet-deflection angle on engine performance, in which $C_{T,v}$ represents the component of vertical thrust divided by the free-stream dynamic pressure $q_0$ and the engine cross-sectional area $A_m$. These data are based on the assumption that nacelle drag does not change with deflection angle.

Altitude. - In the stratosphere (between 35,332 and 105,000 ft), changing flight altitude has only a small effect on propulsive thrust coefficient and specific impulse through the Reynolds number effect on nacelle skin-friction drag coefficient. Below the tropopause, in addition to this Reynolds number effect, the changing ambient temperature significantly affects ram-jet performance. In this region more fuel is required to maintain a design $\tau$ as the altitude is reduced. This

extra mass addition, although it raises the thrust coefficient slightly, lowers the specific impulse considerably. This same increase in fuel consumption is felt by the over-all engine efficiency; however, the increased flight velocity (at constant flight Mach number) reduces the magnitude of the effect. These considerations combine to produce the variations in propulsive thrust coefficient, specific impulse, and over-all efficiency shown in figure 15. Also important is the effect of flight altitude (not shown) on combustion efficiency through changes in ambient pressure and temperature, which in turn establish the temperature, pressure, and velocity at the combustor inlet.

## Off-Design Performance

All the previous discussion has been concerned with a continuously variable-geometry engine or a series of fixed-geometry design-point engines. Thus, it is implied that the inlet capture area is sized to avoid subcritical spillage, the diffuser cone angle is selected for optimum pressure recovery, the diffuser spike can be translated for correct positioning of the cblique shock upon the cowl lip, and the nozzle-throat area and area ratio are optimum. A practical engine incorporating such variable components is not yet available.

This invariance of geometry is of no concern if the engine can always be operated at its design or cruise point. Design-point engine operation is possible for a ram-jet missile that cruises along a Breguet flight path, provided the missile is fully boosted to its cruising Mach number and altitude by some other means. Even after starting cruise flight, however, some corrective action may be required and off-design engine operation may be necessary. Moreover, ram-jet thrust may be desired during the boost phase of the flight. In order to include engine performance for these flight conditions, some off-design engine calculations were also made. Because it was desired to indicate trends rather than absolute magnitudes, a constant value of $\gamma$ of 1.30 for the exhaust gas was used for ease of computation of the off-design performance.

Figure 16 shows the propulsive thrust coefficient and specific impulse of a fixed-geometry engine designed to operate at $M_0$ of 3.5, $M_2$ of 0.200, and $\tau$ of 2.25. These conditions are shown in missile studies to result in good cruise performance. Because the combustor must now operate over a wide range of flight conditions, the combustion efficiency (0.87) was assumed to be slightly lower than that for the design-point case (0.90). Along each line of constant $M_0$, the parameter $\tau$ is raised to increase the thrust coefficient. For any given $\tau$ there is a single unique value of $M_2$ due to the choked fixed-nozzle throat area. At $M_0$ of 3.5 and values of $\tau$ below 2.25, $M_2$ is greater than

0.2 and the diffuser operates supercritically, with a severe loss in pressure recovery and a consequent adverse effect on thrust and specific impulse. As $\tau$ is raised by burning more fuel, $M_2$ is reduced to its design value, and the diffuser then operates critically, with the normal shock positioned at the diffuser lip. This condition corresponds to the sharp break in the curve. Further increase of $\tau$ causes subcritical diffuser operation. Although the external shock or "bow wave" so generated does not necessarily reduce the diffuser pressure recovery severely, it spills air that would normally enter the engine and causes large additive-drag losses. With subcritical operation and no loss in pressure recovery, a small gain in thrust is obtainable over critical operation, but the specific impulse drops markedly. In addition to inefficient operation, subcritical operation often results in instability of flow or "buzzing," which can, in severe cases, even blow out the combustor flame or damage the diffuser structure. In general, then, subcritical operation is undesirable.

For speeds less than design, best engine performance is also generally obtained with the value of $\tau$ chosen to give critical diffuser operation. However, as flight speed is reduced, the value of $M_2$ for critical operation is raised.

The off-design performance of several engines designed for efficient cruising at a flight Mach number of 3.5 and incorporating various types of geometry variation is shown in figure 17 as a function of flight Mach number. Performance is shown for the engines operating at their maximum thrust condition. Also included are data for critical operation of a fixed-geometry engine obtained by cross-plotting the peaks of the curves from figure 16. The performance of an engine with both continuously variable inlet diffuser and exit nozzle is obtained with a wide-open exhaust nozzle and a stoichiometric fuel-air ratio. Extremely large penalties in thrust are suffered with the fixed-geometry engine, with a thrust at flight Mach number 2.5 of only 15 percent of that available from an engine equipped with a continuously variable inlet and outlet. These large thrust losses are mainly due to the necessity of reducing $\tau$ to prevent subcritical operation. The ability to burn more fuel without being forced into the subcritical region explains why the continuously variable engine can produce more thrust than the fixed-geometry engine even at the design Mach number of 3.5.

Equipping an engine with either a movable-spike inlet or a variable-area exit nozzle, both of which are currently feasible, results in considerable gain over fixed-geometry engine performance. With a movable-spike inlet, the spike is translated axially so that all air spillage occurs behind an oblique shock. The flow behind the oblique shock remains supersonic, and the additive drag is not as severe as in the case of a bow wave. This spillage permits $\tau$ to be increased without causing subcritical operation. Thrust increases over the fixed configuration of 50 to 100 percent are possible, but the specific impulse is very low.

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Thrust levels approaching the continuously variable case with about the same specific impulse can be achieved with an engine having a fixed inlet and a movable-plug nozzle. Although the nozzle throat area is variable, the nozzle-exit diameter is fixed and the nozzle expansion ratio cannot be independently chosen. At a flight Mach number of 2.5, the thrust is 74 percent that of the continuously variable engine.

## CONCLUDING REMARKS

The calculated design-point performance of ram-jet engines using JP4 fuel is presented for a wide range of engine total-temperature ratios and combustor-inlet Mach numbers for flight Mach numbers of 1.5 to 4.0. The results, which include engine thrust, drag, fuel consumption, area ratios, and weight, are given in both graphical and tabular form. These data are suitable for use in design studies of ram-jet missiles incorporating either internally or externally mounted engines. Maximum engine specific impulse (including nacelle drag) is approximately 1600 pound-seconds per pound and occurs at a flight Mach number of about 2.5. Over-all engine efficiency, however, continues to increase to a flight Mach number of 4.0. It is not possible on the basis of engine performance data alone to determine best flight Mach number and fuel-air ratio for maximum missile range.

Calculations are also presented which indicate the sensitivity of the design-point results to changes in diffuser pressure recovery, flame-holder pressure loss, combustor length, combustion efficiency, fuel type, nozzle expansion ratio, nozzle velocity coefficient, nozzle jet-deflection angle, and altitude. Significant gains in both thrust coefficient and specific impulse may be achieved by improving the diffuser pressure recovery. However, presently available inlet designs that provide high recovery also have high cowl pressure drags which largely offset this potential gain. Changes in flame-holder pressure loss and combustor length have only small effects on engine performance. These factors may, however, influence combustor efficiency and the resulting missile range. Engine performance is very sensitive to changes in nozzle performance, a 1-percent variation in velocity coefficient often producing a 3-percent variation in engine thrust and specific impulse. Some underexpansion of the exhaust gases is desirable to reduce nacelle drag whenever the nozzle-exit diameter exceeds that of the combustion chamber.

Satisfactory off-design operation of a ram-jet engine is not possible with a fixed-geometry configuration. Somewhat better thrust can be obtained with the added complication of a translating-spike diffuser, although the specific impulse is poorer. Use of a movable plug to vary

the throat area of the nozzle yields both thrust and specific impulse approaching those of a continuously variable-geometry engine. In considering engines designed for good cruise performance at flight Mach number 3.5 but operating at flight Mach number 2.5, the maximum thrusts are 15, 37, and 74 percent of that available with continuously variable geometry for engines with fixed-geometry, translating-spike diffuser, and movable-plug nozzle, respectively.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, August 5, 1954

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## APPENDIX A

## SYMBOLS

The following symbols are used in this report:

| | |
|---|---|
| A | area, sq ft |
| $A_m$ | diffuser capture area $A_1$ or combustion-chamber area $A_3$, whichever is larger, sq ft |
| $C_D$ | nacelle drag coefficient, $D/q_0A_m$ |
| $C_F$ | net thrust coefficient, $F/q_0A_m$ |
| $C_T$ | propulsive thrust coefficient, $C_F - C_D$ |
| $C_V$ | nozzle velocity coefficient |
| D | drag, lb |
| d | diameter, ft |
| E | over-all engine efficiency, $(F - D)V_0/JH\ w_f$ |
| F | net thrust, $m_6V_6 - m_0V_0 + A_6(p_6 - p_0)$, lb |
| f/a | fuel-air ratio |
| H | lower heating value of fuel, Btu/lb |
| I | fuel specific impulse, $(F - D)/w_f$, lb-sec/lb |
| $I_F$ | fuel specific impulse not including drag, $F/w_f$, lb-sec/lb |
| J | mechanical equivalent of heat, 778 ft-lb/Btu |
| $l$ | combustion-chamber length, ft |
| M | Mach number |
| m | mass-flow rate, slugs/sec |
| P | total pressure, lb/sq ft |
| p | static pressure, lb/sq ft |

[redacted]

| | |
|---|---|
| $q$ | incompressible dynamic pressure, $\rho V^2/2$, lb/sq ft |
| $R$ | gas constant, 53.4 ft-lb/(lb)($^\circ$R) |
| $T$ | total temperature, $^\circ$R |
| $t$ | static temperature, $^\circ$R |
| $V$ | velocity, ft/sec |
| $w_f$ | fuel-flow rate, lb/sec |
| $\gamma$ | ratio of specific heat at constant pressure to specific heat at constant volume |
| $\eta_c$ | combustion efficiency, $w_{f,id}/w_{f,ac}$ |
| $\lambda$ | jet-deflection angle, deg |
| $\rho$ | density, lb/cu ft |
| $\tau$ | engine total-temperature ratio, $T_4/T_3$ |

Subscripts:

| | |
|---|---|
| ac | actual |
| ef | effective |
| fr | friction |
| id | ideal |
| nom | nominal |
| v | vertical |
| 0 | free stream |
| 1 | diffuser inlet |
| 2 | combustion-chamber inlet (upstream of flame holder) |
| 3 | combustion-chamber inlet (downstream of flame holder) |
| 4 | combustion-chamber exit |
| 5 | nozzle throat |
| 6 | nozzle exit |

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## APPENDIX B

## THERMODYNAMIC ASSUMPTIONS

Assigning correct gas properties and maintaining high computational accuracy are very important in any ram-jet engine analysis, because small changes in the calculated jet thrust may be magnified 3 or 4 times in the net propulsive thrust. Specification of the gas properties involves realistic choice of $\gamma$, R, and $\tau$ as a function of temperature, fuel-air ratio, and pressure (if there is appreciable dissociation).

Data for octane (ref. 4), which include dissociation effects, were converted to JP4 and used in constructing figure 3, which gives $\tau$ as a function of $M_0$ and f/a for an ambient temperature of 392° R. Similar curves were constructed for altitudes under the tropopause where the ambient temperature is different from 392° R. The very high temperatures plus the changes of composition due to burning cause the $\gamma$ of the combustion gases to vary appreciably from 1.40 if equilibrium is reached. These equilibrium values, as a function of temperature and f/a and for a pressure of 1 atmosphere, were obtained from reference 10, which includes a very appreciable effect of dissociation. Molecular equilibrium is not necessarily maintained during the expansion of the gas through the nozzle. However, considerations of heat-capacity lag and chemical-reaction rates indicated that the process is probably closer to equilibrium than to "frozen" conditions. After trying several different methods, it was found that specifying an effective $\gamma$ for the nozzle by

$$\gamma_{ef} = \frac{1}{2}(\gamma_4 + \gamma_6)$$

gave best results as checked by equilibrium calculations using references 11 and 12. The deviations in net thrust were generally found to be less than 3 percent. According to reference 10, this means of specifying $\gamma_{ef}$ is also best for use in the isentropic equation

$$\frac{t}{T} = \left(\frac{p}{P}\right)^{\frac{\gamma-1}{\gamma}}$$

The value of R for the air stream was taken as 53.4 ft-lb/(lb)(°R). The same value was used for the exhaust gas with small error (ref. 13).

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## APPENDIX C

## ENGINE WEIGHT

The engine was assumed to be constructed entirely of stainless steel with a density of 490 pounds per cubic foot of material. The allowable stress was 100,000 psi in the diffuser and 50,000 psi in the combustion chamber and nozzle. The engine was designed to withstand a pressure of 110 psig, which would be obtained at $M_0 = 3.5$ and an altitude of 35,000 feet. The diffuser island was made the same gage as the diffuser shell and the nozzle the same as the combustion-chamber wall. The external nacelle thickness was taken as 0.020 inch. An allowance of 50 pounds per square foot of combustion-chamber area was made for fuel systems and controls. After making arbitrary allowances for stiffeners and supports that were felt to be conservative, the weight of typical engines designed for several flight speeds was calculated. In all cases the calculated weight was very nearly 180 pounds per square foot of combustor cross-sectional area. Because missile studies show that ram-jet engine weight is of secondary importance, it was assumed that the value of 180 pounds per square foot was sufficiently accurate for all engines considered.

## APPENDIX D

## ENGINE DRAG

The engine thrust is defined in terms of the total-momentum changes occurring between the free-stream tube ahead of the engine and the exhaust at the end of the engine. The drag must, therefore, include all the external forces acting on the stream tube and engine between the same two points. These forces are composed of pressure forces acting perpendicular to the stream tube and engine, and a friction force acting parallel to the engine nacelle due to the viscosity of the air. The pressure (or wave) drag may be further broken down into the part acting directly on the nacelle and the additional, imaginary part acting on the stream tube because of the way the thrust was defined. The latter force is termed additive drag, and was calculated by the method of reference 14.

The nacelle friction drag was calculated from the following equation, which is based on the flat-plate formula of reference 15:

$$C_{D,fr} = \frac{A_W}{A_3} \frac{0.0306\ K}{Re^{1/7}\left(1 + \frac{\gamma-1}{4} M_0^2\right)^{5/7}}$$

where $A_W$ is nacelle skin area, K is a shape factor taken as 1.05 for a cylindrical nacelle, and the Reynolds number Re is based on free-stream conditions with a nominal length of 30 feet and a nominal altitude of 70,000 feet.

The nacelle pressure drag was obtained from the linearized theory from reference 16. For the low-speed cases when the exit area was smaller than combustion-chamber area, data for boattail drag for a $7.04^\circ$ cone were used from the same source. The diffuser cowl was assumed conical, with no added pressure drag incurred from a curved lip.

The engine was assumed to have a fineness ratio (length divided by combustion-chamber diam.) of 9, the diffuser being nominally 4, combustion chamber 3, and nozzle 2. The appearance of the engine as a function of flight speed is as shown in the following sketch (not to scale):

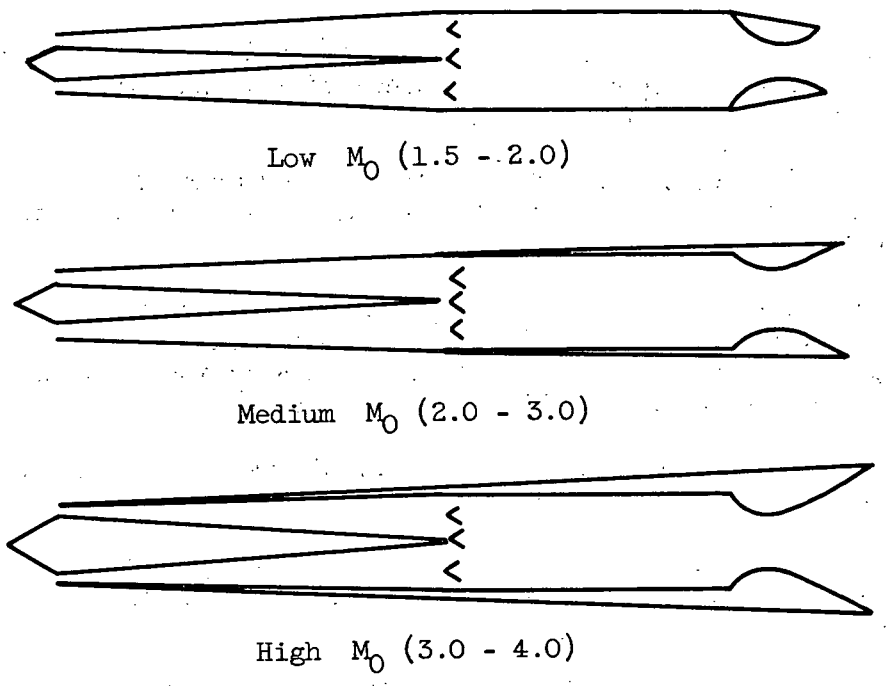


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TABLE I. - DESIGN-POINT RAM-JET ENGINE PERFORMANCE

| Flight Mach number, $M_0$ | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 1.5 | | | | | | 2.0 | | | | | |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.150 | | | | | | | | | | | |
| $A_1/A_3$ = 0.287 | | | | | | $A_1/A_3$ = 0.382 | | | | | |
| $\tau$ | $C_T$ | I | $C_D$ | $A_6/A_3$ | $A_5/A_3$ | $\tau$ | $C_T$ | I | $C_D$ | $A_6/A_3$ | $A_5/A_3$ |
| 2.0 | 0.023 | 152 | 0.186 | 0.468 | 0.395 | 2.5 | 0.291 | 1274 | 0.118 | 0.750 | 0.450 |
| 3.0 | .210 | 907 | .175 | .585 | .499 | 3.0 | .428 | 1372 | .009 | .840 | .499 |
| 4.0 | .383 | 1054 | .161 | .699 | .590 | 3.5 | .551 | 1400 | .098 | .925 | .548 |
| 5.0 | .533 | 1080 | .143 | .812 | .690 | 4.0 | .670 | 1393 | .086 | 1.005 | .595 |
| 6.0 | .698 | 1056 | .122 | .931 | .794 | 4.5 | .780 | 1360 | .087 | 1.085 | .643 |
| 7.0 | .882 | 998 | .109 | 1.060 | .909 | 5.0 | .880 | 1307 | .088 | 1.174 | .696 |
| 7.32 | .945 | 975 | .109 | 1.106 | .945 | 5.5 | .990 | 1227 | .090 | 1.275 | .751 |
| | | | | | | 5.96 | 1.096 | 1139 | .092 | 1.380 | .807 |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.175 | | | | | | | | | | | |
| $A_1/A_3$ = 0.334 | | | | | | $A_1/A_3$ = 0.444 | | | | | |
| 2.0 | 0.055 | 418 | 0.175 | 0.550 | 0.465 | 2.5 | 0.365 | 1426 | 0.104 | 0.874 | 0.530 |
| 3.0 | .282 | 1043 | .158 | .694 | .590 | 3.0 | .524 | 1508 | .086 | .985 | .593 |
| 4.0 | .471 | 1139 | .138 | .825 | .713 | 3.5 | .660 | 1488 | .084 | 1.086 | .654 |
| 5.0 | .635 | 1106 | .113 | .970 | .843 | 4.0 | .780 | 1438 | .085 | 1.180 | .720 |
| *6.0 | .780 | 984 | .109 | 1.124 | 1.000 | 4.5 | .895 | 1369 | .087 | 1.287 | .790 |
| | | | | | | 5.0 | 1.007 | 1288 | .089 | 1.400 | .863 |
| | | | | | | 5.5 | 1.122 | 1197 | .092 | 1.527 | .940 |
| | | | | | | *5.85 | 1.206 | 1130 | .094 | 1.618 | 1.000 |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.200 | | | | | | | | | | | |
| $A_1/A_3$ = 0.379 | | | | | | $A_1/A_3$ = 0.505 | | | | | |
| 2.5 | 0.225 | 996 | 0.150 | 0.726 | 0.620 | 2.25 | 0.347 | 1433 | 0.091 | 0.941 | 0.581 |
| 3.0 | .344 | 1131 | .137 | .808 | .697 | 2.5 | .440 | 1498 | .081 | 1.009 | .620 |
| 3.5 | .454 | 1177 | .121 | .896 | .779 | 2.75 | .520 | 1527 | .082 | 1.072 | .660 |
| 4.0 | .550 | 1178 | .103 | .986 | .865 | 3.0 | .598 | 1519 | .083 | 1.139 | .700 |
| 4.5 | .625 | 1136 | .100 | 1.080 | .952 | 3.25 | .672 | 1502 | .084 | 1.194 | .743 |
| *4.77 | .659 | 1065 | .101 | 1.132 | 1.000 | 3.75 | .804 | 1448 | .087 | 1.324 | .832 |
| | | | | | | 4.25 | .927 | 1371 | .090 | 1.460 | .929 |
| | | | | | | *4.60 | 1.006 | 1312 | .093 | 1.560 | 1.000 |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.225 | | | | | | | | | | | |
| $A_1/A_3$ = 0.424 | | | | | | $A_1/A_3$ = 0.564 | | | | | |
| 2.50 | 0.266 | 1079 | 0.130 | 0.835 | 0.719 | 2.25 | 0.399 | 1469 | 0.081 | 1.078 | 0.675 |
| 2.75 | .348 | 1151 | .122 | .876 | .766 | 2.50 | .488 | 1492 | .083 | 1.152 | .721 |
| 3.00 | .400 | 1180 | .111 | .931 | .819 | 2.75 | .574 | 1497 | .084 | 1.226 | .770 |
| 3.25 | .452 | 1177 | .099 | .988 | .870 | 3.00 | .657 | 1481 | .086 | 1.302 | .822 |
| 3.50 | .499 | 1144 | .097 | 1.040 | .928 | 3.25 | .733 | 1457 | .087 | 1.378 | .879 |
| *3.80 | .540 | 1047 | .099 | 1.110 | 1.000 | 3.50 | .801 | 1423 | .089 | 1.456 | .940 |
| | | | | | | *3.72 | .863 | 1377 | .091 | 1.530 | 1.000 |

*Thermal choking

[redacted]

TABLE I. - Continued. DESIGN-POINT RAM-JET ENGINE PERFORMANCE

| Flight Mach number, $M_0$ | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 2.5 | | | | | | 3.0 | | | | | |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.150 | | | | | | | | | | | |
| $A_1/A_3$ = 0.505 | | | | | | $A_1/A_3$ = 0.626 | | | | | |
| $\tau$ | $C_T$ | I | $C_D$ | $A_6/A_3$ | $A_5/A_3$ | $\tau$ | $C_T$ | I | $C_D$ | $A_6/A_3$ | $A_5/A_3$ |
| 2.0 | 0.284 | 1441 | 0.076 | 0.960 | 0.397 | 2.00 | 0.390 | 1522 | 0.061 | 1.177 | 0.398 |
| 2.5 | .469 | 1555 | .072 | 1.052 | .453 | 2.25 | .507 | 1565 | .062 | 1.235 | .429 |
| 3.0 | .638 | 1546 | .073 | 1.125 | .501 | 2.50 | .620 | 1566 | .063 | 1.290 | .456 |
| 3.5 | .795 | 1510 | .074 | 1.193 | .552 | 2.75 | .730 | 1550 | .064 | 1.344 | .481 |
| 4.0 | .960 | 1432 | .075 | 1.275 | .606 | 3.00 | .834 | 1516 | .065 | 1.402 | .506 |
| 4.5 | 1.132 | 1329 | .077 | 1.371 | .661 | 3.25 | .951 | 1474 | .066 | 1.466 | .537 |
| 4.80 | 1.250 | 1249 | .078 | 1.436 | .697 | 3.50 | 1.079 | 1419 | .069 | 1.540 | .559 |
| | | | | | | 3.96 | 1.349 | 1300 | .073 | 1.701 | .619 |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.175 | | | | | | | | | | | |
| $A_1/A_3$ = 0.586 | | | | | | $A_1/A_3$ = 0.727 | | | | | |
| 2.0 | 0.348 | 1512 | 0.070 | 1.062 | 0.468 | 2.00 | 0.458 | 1537 | 0.058 | 1.305 | 0.472 |
| 2.5 | .554 | 1594 | .071 | 1.138 | .534 | 2.25 | .591 | 1576 | .061 | 1.370 | .507 |
| 3.0 | .749 | 1565 | .073 | 1.246 | .598 | 2.50 | .722 | 1574 | .063 | 1.430 | .538 |
| 3.5 | .934 | 1508 | .075 | 1.338 | .665 | 2.75 | .851 | 1551 | .066 | 1.499 | .571 |
| 4.0 | 1.125 | 1406 | .077 | 1.432 | .734 | 3.00 | .972 | 1515 | .067 | 1.567 | .606 |
| 4.5 | 1.302 | 1299 | .079 | 1.535 | .810 | 3.25 | 1.105 | 1462 | .069 | 1.644 | .641 |
| 4.80 | 1.420 | 1212 | .081 | 1.600 | .860 | 3.50 | 1.249 | 1402 | .072 | 1.732 | .674 |
| | | | | | | 3.96 | 1.546 | 1266 | .078 | 1.942 | .747 |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.200 | | | | | | | | | | | |
| $A_1/A_3$ = 0.666 | | | | | | $A_1/A_3$ = 0.826 | | | | | |
| 2.0 | 0.400 | 1537 | 0.071 | 1.158 | 0.545 | 2.00 | 0.520 | 1531 | 0.065 | 1.425 | 0.545 |
| 2.5 | .630 | 1597 | .072 | 1.262 | .625 | 2.25 | .657 | 1561 | .067 | 1.502 | .590 |
| 3.0 | .850 | 1566 | .074 | 1.363 | .706 | 2.50 | .801 | 1555 | .068 | 1.578 | .633 |
| 3.5 | 1.052 | 1501 | .076 | 1.470 | .793 | 2.75 | .953 | 1533 | .070 | 1.652 | .673 |
| 4.0 | 1.250 | 1396 | .079 | 1.589 | .888 | 3.00 | 1.091 | 1496 | .072 | 1.727 | .714 |
| *4.45 | 1.420 | 1287 | .082 | 1.710 | 1.000 | 3.25 | 1.239 | 1444 | .074 | 1.822 | .763 |
| | | | | | | 3.50 | 1.390 | 1375 | .076 | 1.908 | .811 |
| | | | | | | 3.96 | 1.692 | 1220 | .084 | 2.162 | .918 |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.225 | | | | | | | | | | | |
| $A_1/A_3$ = 0.745 | | | | | | $A_1/A_3$ = 0.923 | | | | | |
| 2.00 | 0.439 | 1519 | 0.071 | 1.240 | 0.625 | 2.00 | 0.589 | 1551 | 0.053 | 1.553 | 0.629 |
| 2.25 | .565 | 1556 | .073 | 1.301 | .675 | 2.25 | .748 | 1556 | .055 | 1.628 | .680 |
| 2.50 | .689 | 1565 | .074 | 1.360 | .722 | 2.50 | .900 | 1544 | .072 | 1.714 | .730 |
| 2.75 | .810 | 1555 | .075 | 1.421 | .775 | 2.75 | 1.052 | 1513 | .074 | 1.817 | .787 |
| 3.00 | .920 | 1527 | .076 | 1.486 | .829 | 3.00 | 1.197 | 1465 | .076 | 1.901 | .840 |
| 3.25 | 1.028 | 1487 | .077 | 1.551 | .897 | 3.25 | 1.351 | 1410 | .078 | 2.022 | .901 |
| *3.66 | 1.201 | 1400 | .080 | 1.669 | 1.000 | *3.58 | 1.584 | 1314 | .083 | 2.168 | 1.000 |

*Thermal choking

[redacted]

TABLE I. - Concluded. DESIGN-POINT RAM-JET ENGINE PERFORMANCE

| Flight Mach number, $M_0$ | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 3.5 | | | | | | 4.0 | | | | | |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.125 | | | | | | | | | | | |
| $A_1/A_3$ = 0.618 | | | | | | $A_1/A_3$ = 0.743 | | | | | |
| $\tau$ | $C_T$ | I | $C_D$ | $A_6/A_3$ | $A_5/A_3$ | $\tau$ | $C_T$ | I | $C_D$ | $A_6/A_3$ | $A_5/A_3$ |
| 1.75 | 0.271 | 1349 | 0.055 | 1.209 | 0.303 | 1.50 | 0.162 | 954 | 0.047 | 1.362 | 0.281 |
| 2.00 | .388 | 1411 | .056 | 1.275 | .329 | 1.75 | .334 | 1255 | .050 | 1.457 | .308 |
| 2.25 | .505 | 1428 | .057 | 1.341 | .355 | 2.00 | .493 | 1324 | .052 | 1.543 | .334 |
| 2.50 | .628 | 1422 | .058 | 1.406 | .376 | 2.25 | .654 | 1334 | .054 | 1.627 | .356 |
| 2.75 | .759 | 1398 | .059 | 1.488 | .400 | 2.50 | .838 | 1310 | .058 | 1.804 | .383 |
| 3.00 | .900 | 1353 | .062 | 1.615 | .426 | 2.75 | 1.047 | 1218 | .064 | 2.048 | .412 |
| 3.25 | 1.059 | 1276 | .064 | 1.744 | .451 | | | | | | |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.150 | | | | | | | | | | | |
| $A_1/A_3$ = 0.740 | | | | | | $A_1/A_3$ = 0.890 | | | | | |
| 1.75 | 0.341 | 1375 | 0.054 | 1.358 | 0.370 | 1.50 | 0.213 | 1045 | 0.049 | 1.536 | 0.399 |
| 2.00 | .470 | 1441 | .056 | 1.437 | .401 | 1.75 | .403 | 1263 | .052 | 1.653 | .373 |
| 2.25 | .617 | 1450 | .058 | 1.515 | .430 | 2.00 | .592 | 1329 | .055 | 1.762 | .403 |
| 2.50 | .755 | 1442 | .060 | 1.606 | .459 | 2.25 | .784 | 1329 | .058 | 1.880 | .435 |
| 2.75 | .908 | 1412 | .062 | 1.695 | .487 | 2.50 | 1.001 | 1314 | .063 | 2.086 | .466 |
| 3.00 | 1.076 | 1360 | .066 | 1.856 | .519 | 2.75 | 1.254 | 1217 | .071 | 2.356 | .501 |
| 3.25 | 1.255 | 1266 | .070 | 2.015 | .549 | | | | | | |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.175 | | | | | | | | | | | |
| $A_1/A_3$ = 0.858 | | | | | | $A_1/A_3$ = 1.032 | | | | | |
| 1.75 | 0.389 | 1410 | 0.058 | 1.524 | 0.438 | 1.50 | 0.244 | 1035 | 0.055 | 1.730 | 0.401 |
| 2.00 | .553 | 1452 | .060 | 1.609 | .474 | 1.75 | .467 | 1263 | .057 | 1.844 | .438 |
| 2.25 | .720 | 1462 | .062 | 1.691 | .507 | 2.00 | .687 | 1329 | .061 | 1.986 | .477 |
| 2.50 | .887 | 1450 | .064 | 1.805 | .544 | 2.25 | .906 | 1335 | .064 | 2.125 | .513 |
| 2.75 | 1.060 | 1415 | .067 | 1.912 | .578 | 2.50 | 1.158 | 1310 | .069 | 2.352 | .551 |
| 3.00 | 1.240 | 1353 | .072 | 2.104 | .620 | 2.75 | 1.449 | 1213 | .078 | 2.680 | .598 |
| 3.25 | 1.440 | 1250 | .077 | 2.284 | .662 | | | | | | |
| Combustion-chamber-inlet Mach number, $M_2$ = 0.200 | | | | | | | | | | | |
| $A_1/A_3$ = 0.976 | | | | | | | | | | | |
| 1.75 | 0.428 | 1375 | 0.063 | 1.665 | 0.506 | | | | | | |
| 2.00 | .610 | 1420 | .065 | 1.768 | .551 | | | | | | |
| 2.25 | .800 | 1436 | .067 | 1.871 | .594 | | | | | | |
| 2.50 | .995 | 1426 | .069 | 1.991 | .638 | | | | | | |
| 2.75 | 1.195 | 1394 | .072 | 2.118 | .683 | | | | | | |
| 3.00 | 1.400 | 1340 | .077 | 2.311 | .732 | | | | | | |
| 3.25 | 1.634 | 1240 | .084 | 2.560 | .790 | | | | | | |

[redacted]

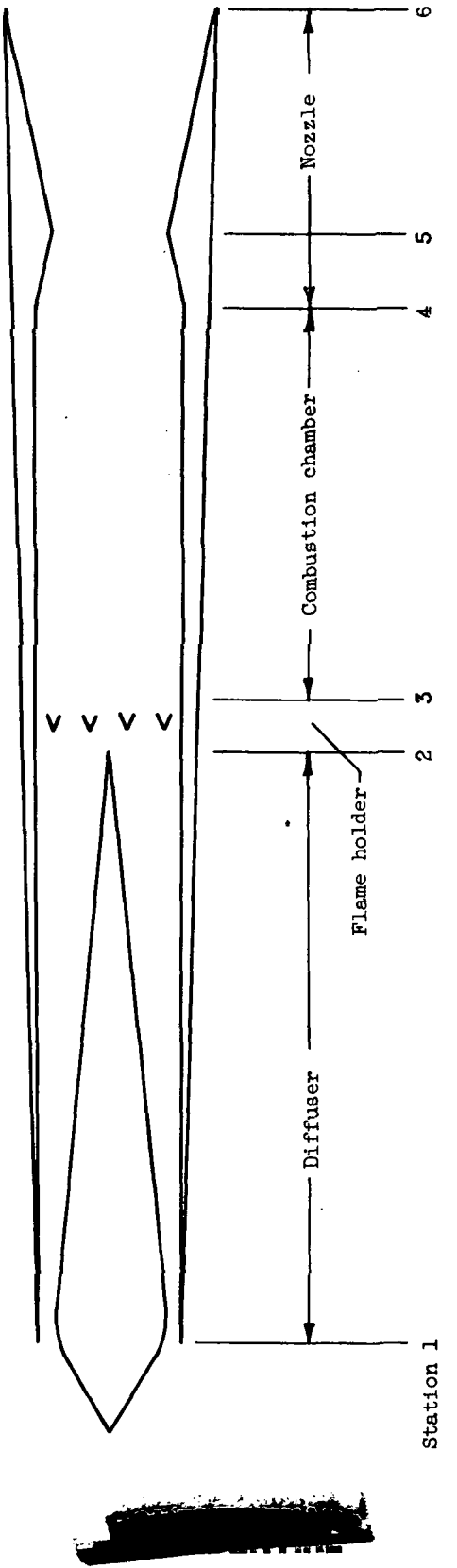



Figure 1. - Schematic diagram of typical ram-jet engine.

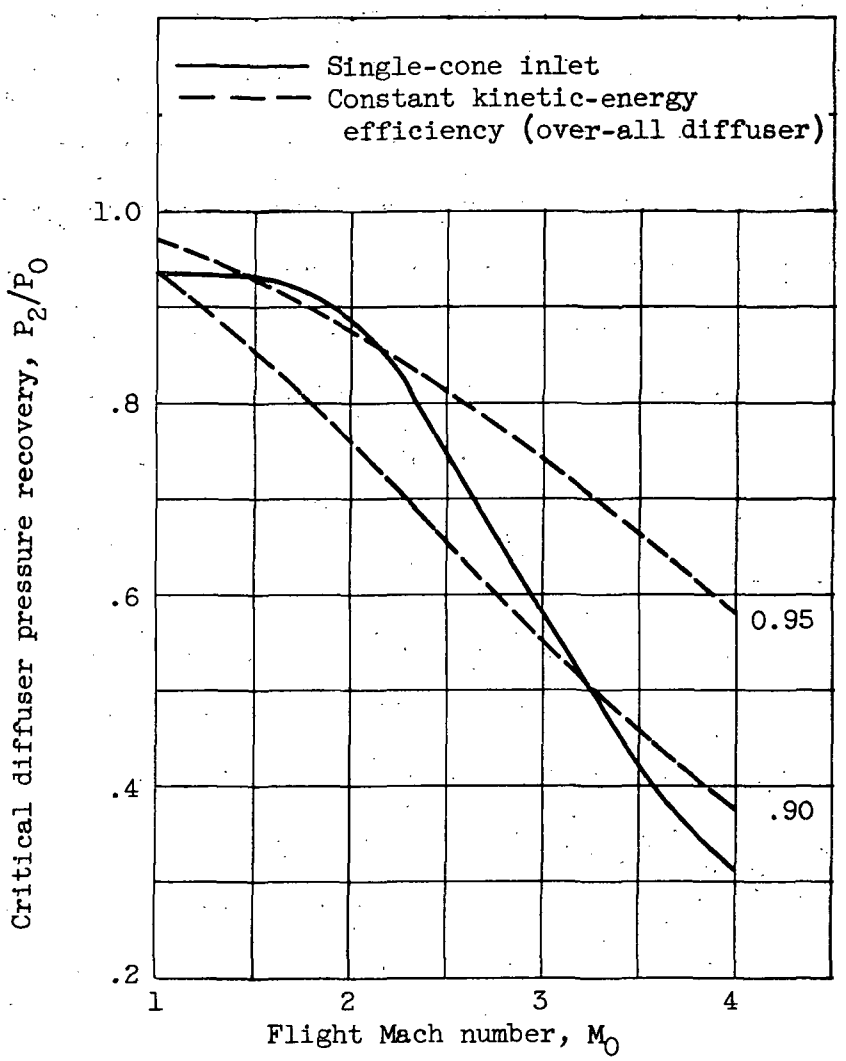



Figure 2. - Effect of flight Mach number on critical diffuser pressure recovery. Single-cone spike inlet.

[redacted]

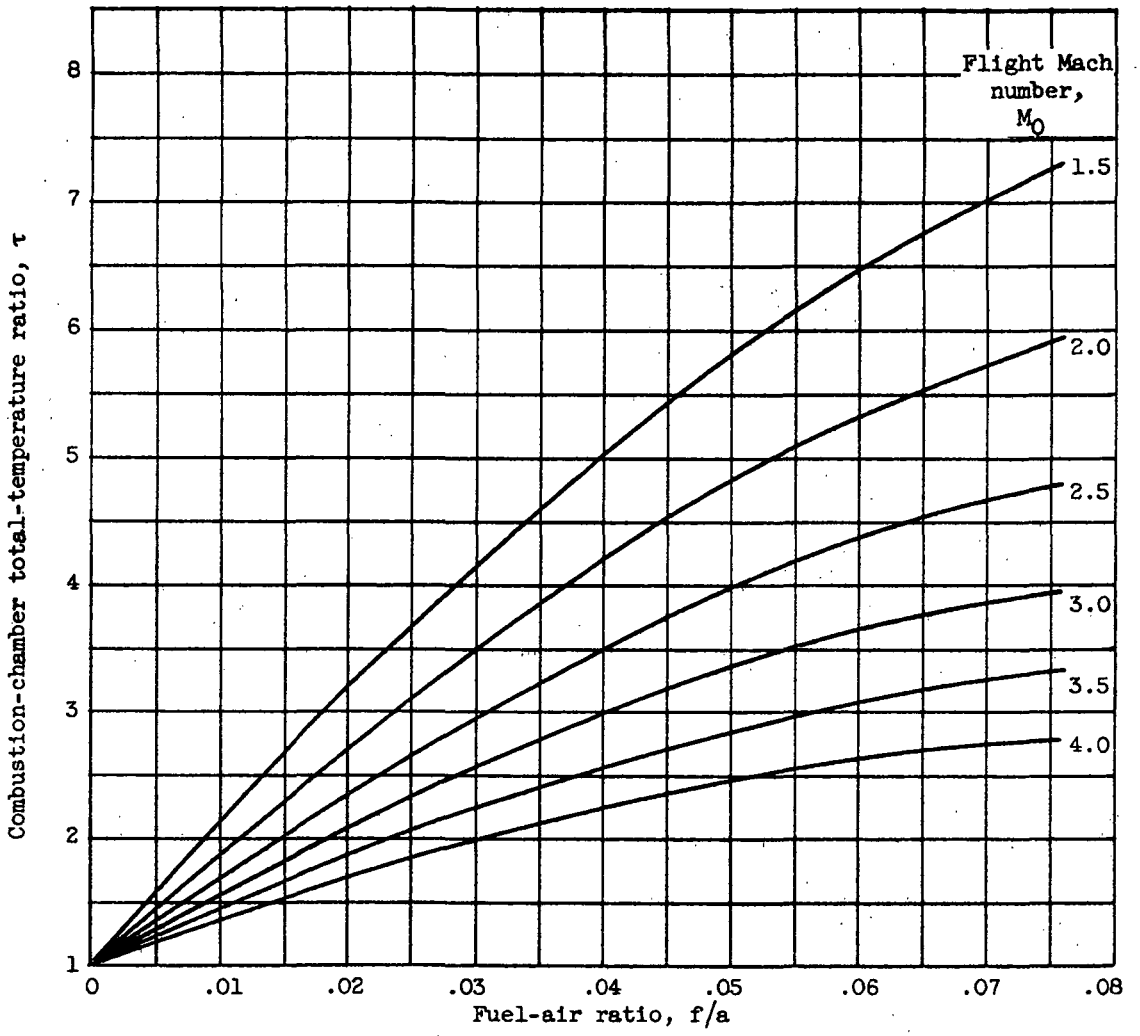
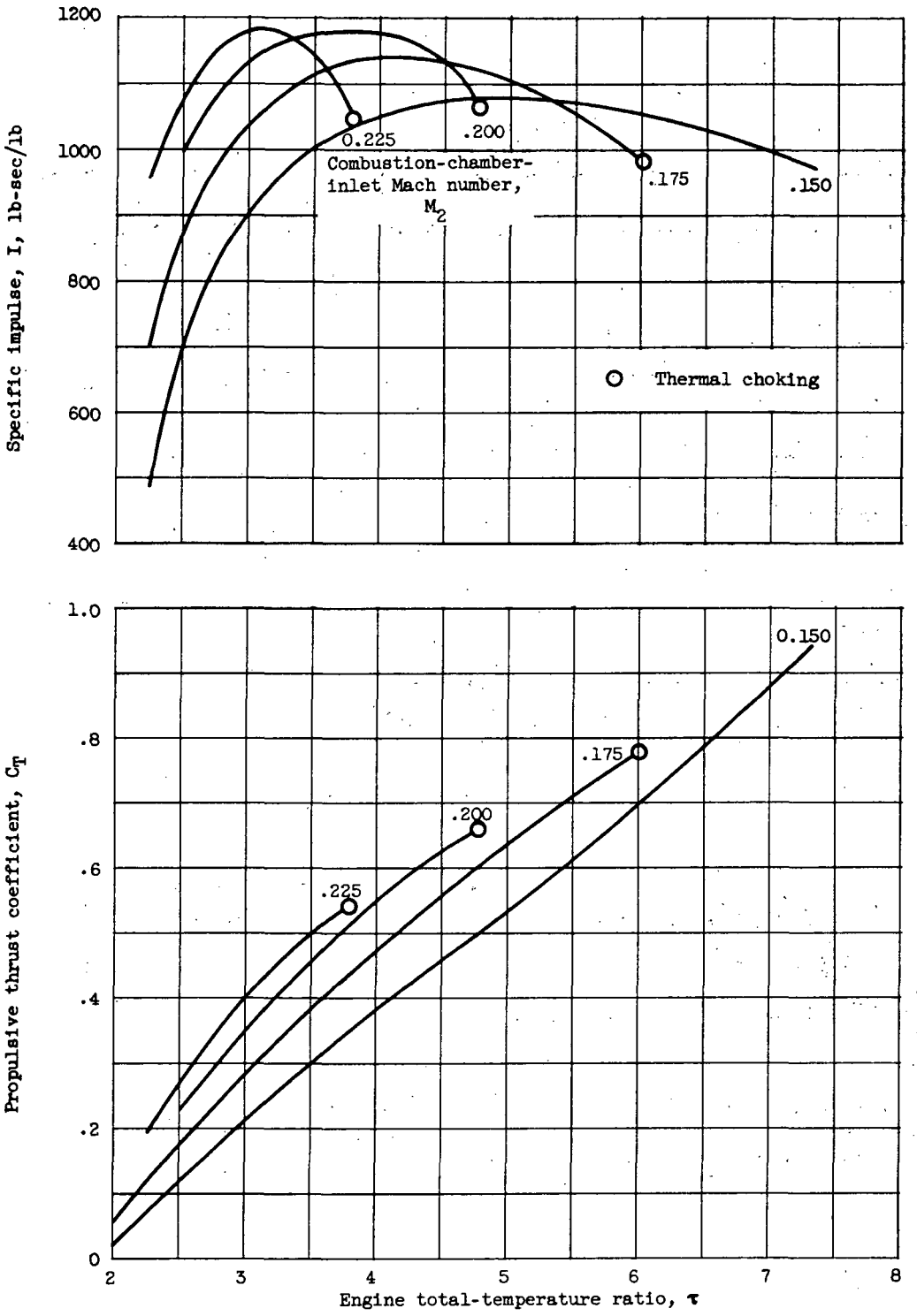


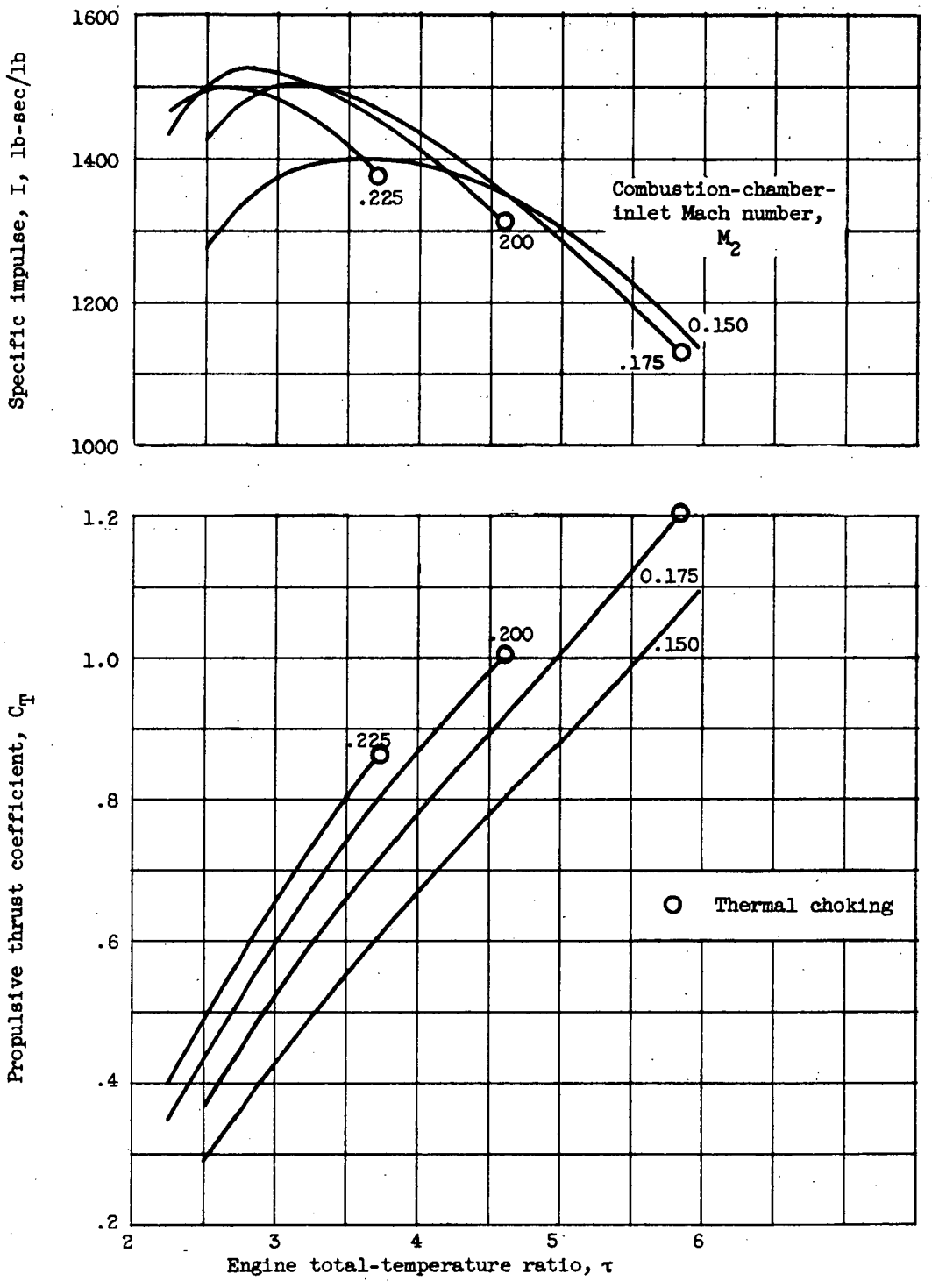

Figure 3. - Effect of fuel-air ratio on combustion-chamber total-temperature ratio for JP4 fuel (heating value, 18,640 Btu/lb). Ambient temperature, 392° R; combustion efficiency, 0.90.




(a) Flight Mach number, 1.5.

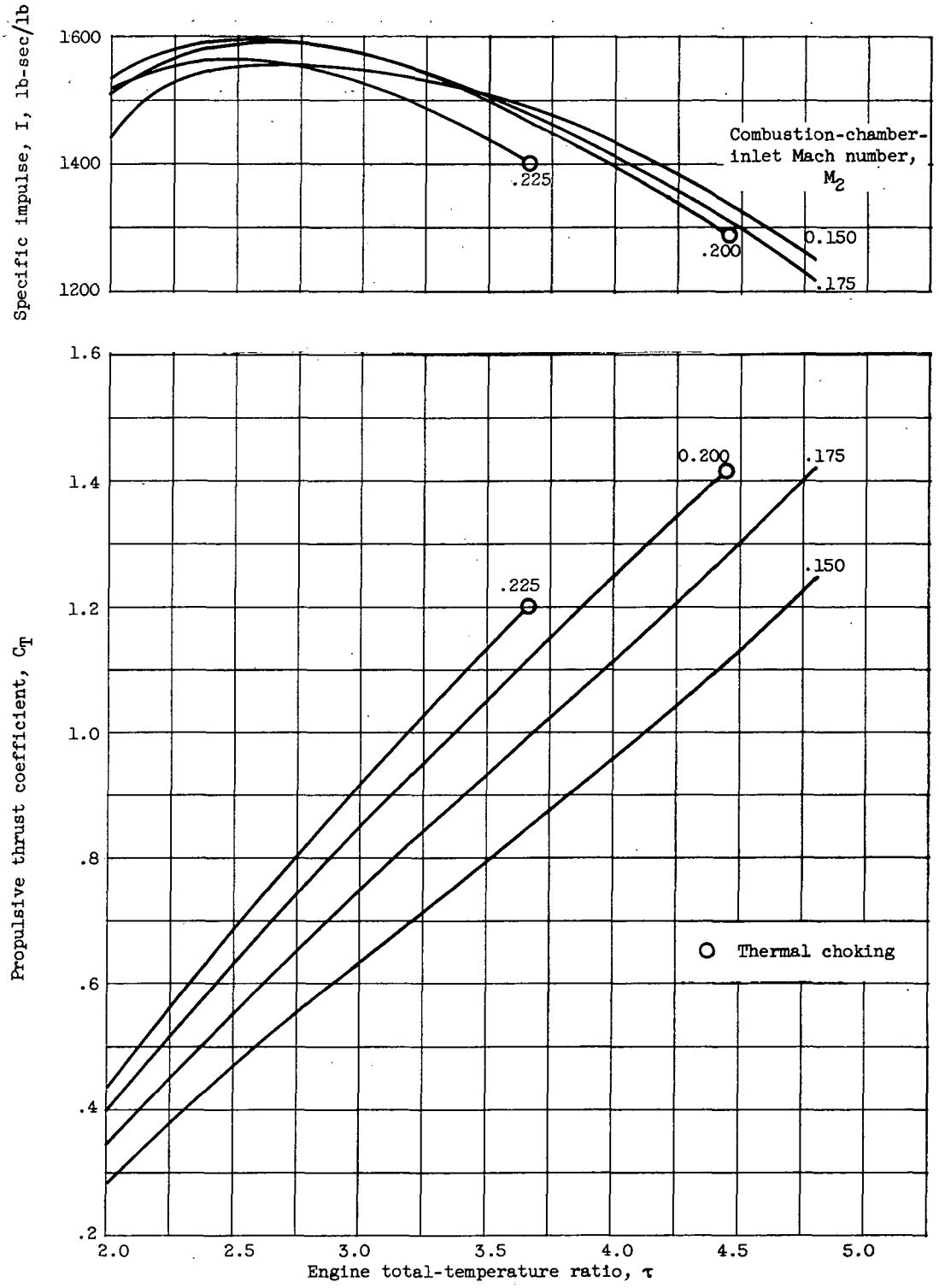
Figure 4. - Design-point performance of ram-jet engine.

[redacted]




(b) Flight Mach number, 2.0.

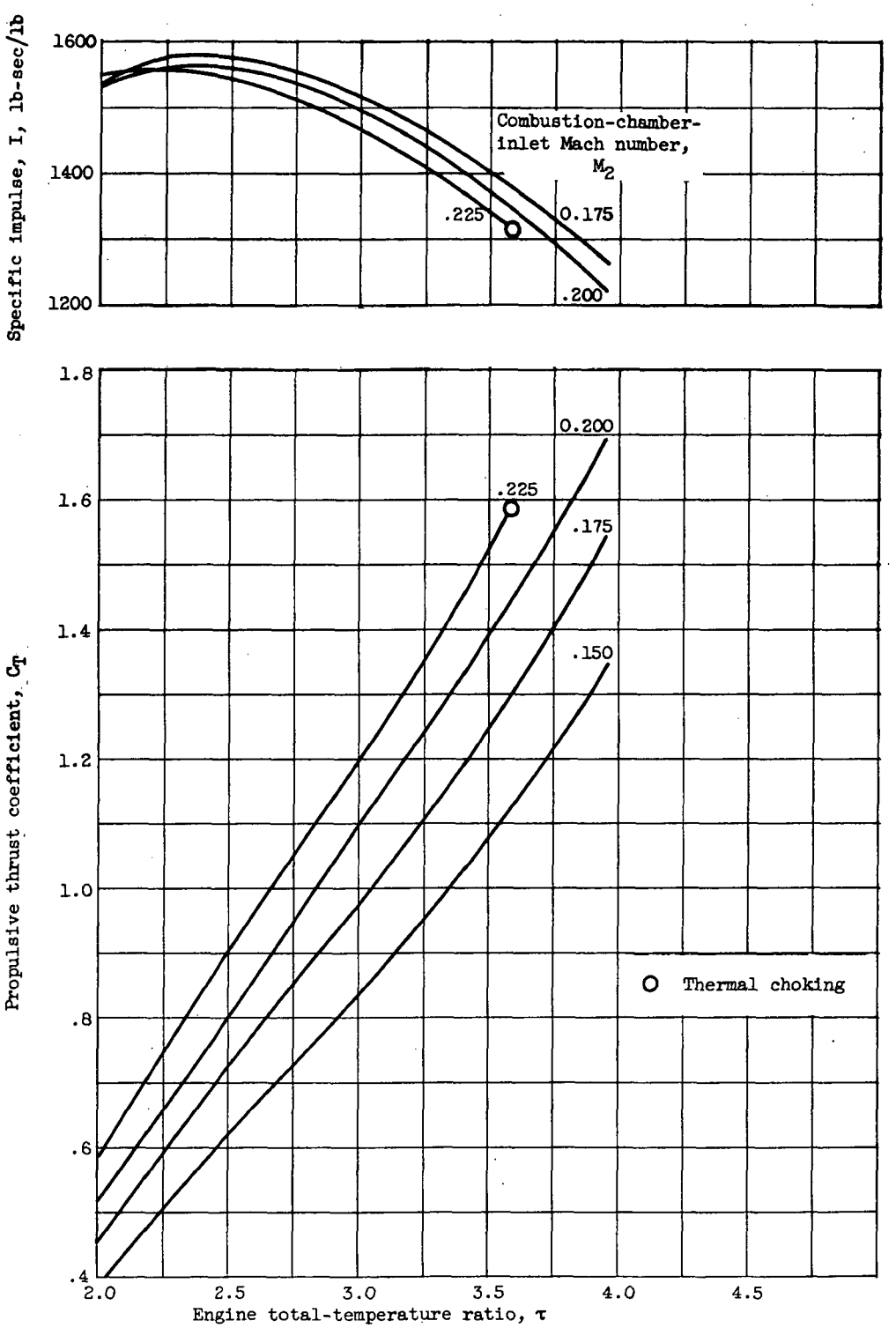
Figure 4. - Continued. Design-point performance of ram-jet engine.




(c) Flight Mach number, 2.5.

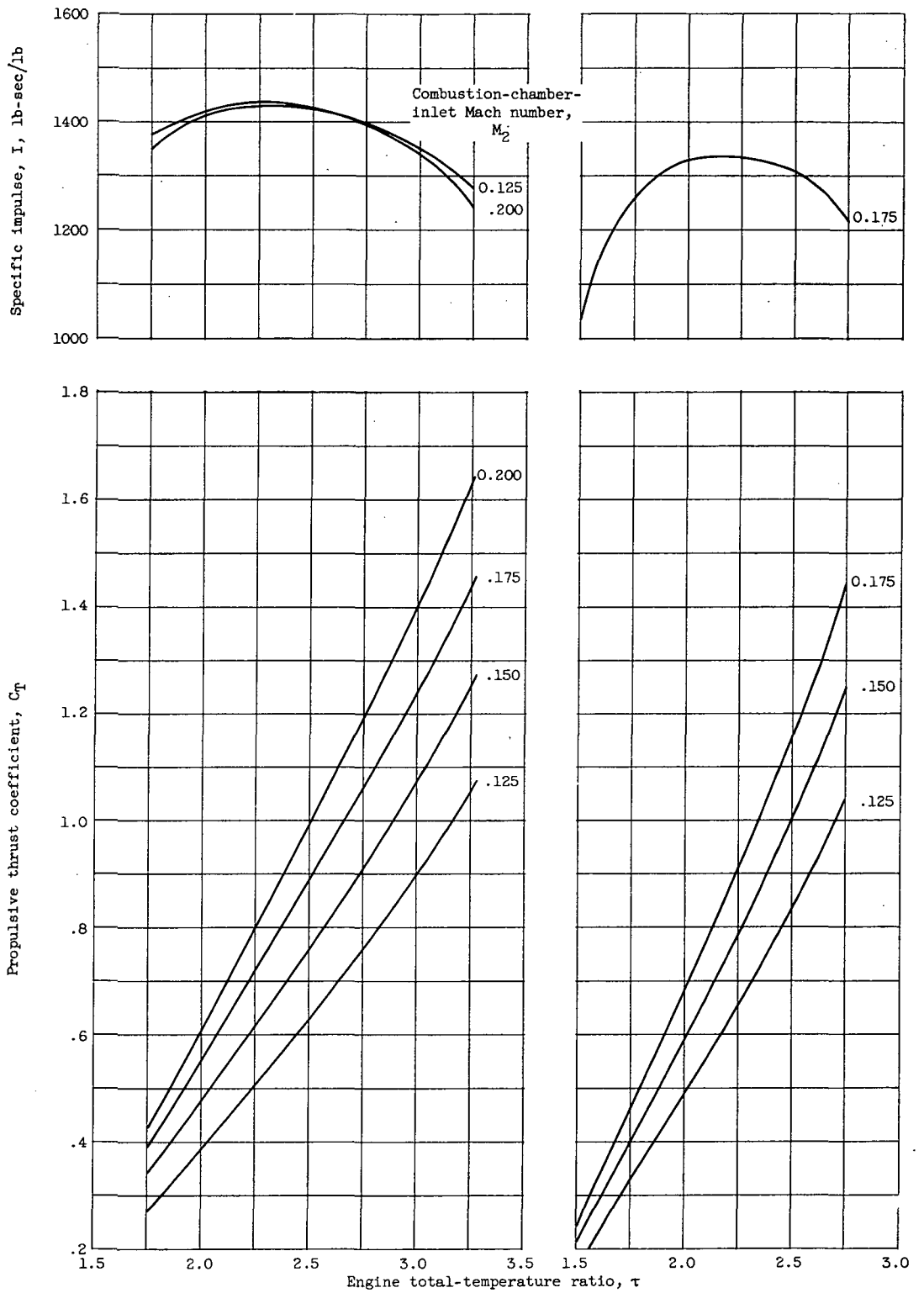
Figure 4. - Continued. Design-point performance of ram-jet engine.

[redacted]




(d) Flight Mach number, 3.0.

Figure 4. - Continued. Design-point performance of ram-jet engine.




(e) Flight Mach number, 3.5. (f) Flight Mach number, 4.0.

Figure 4. - Concluded. Design-point performance of ram-jet engine.

[redacted]

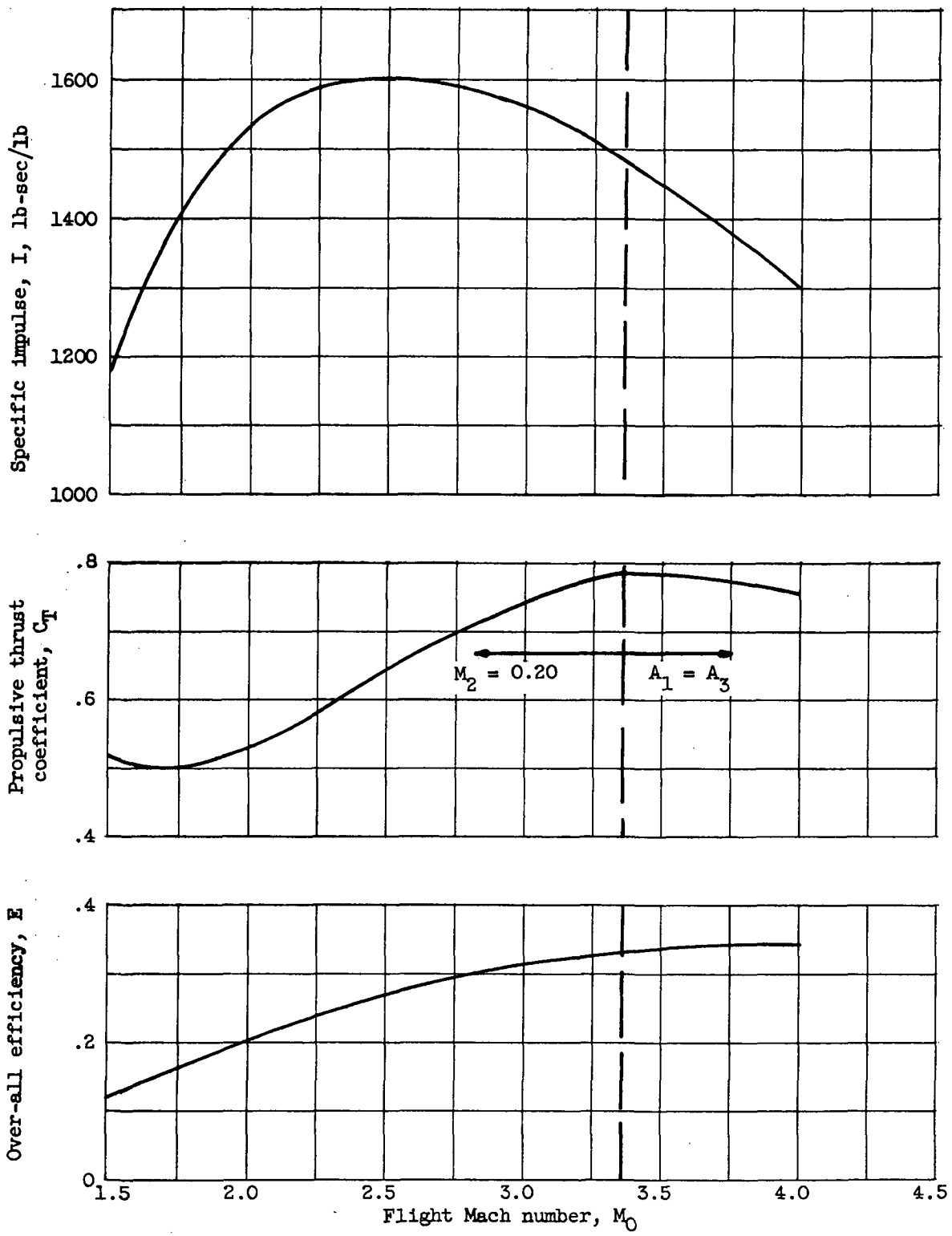
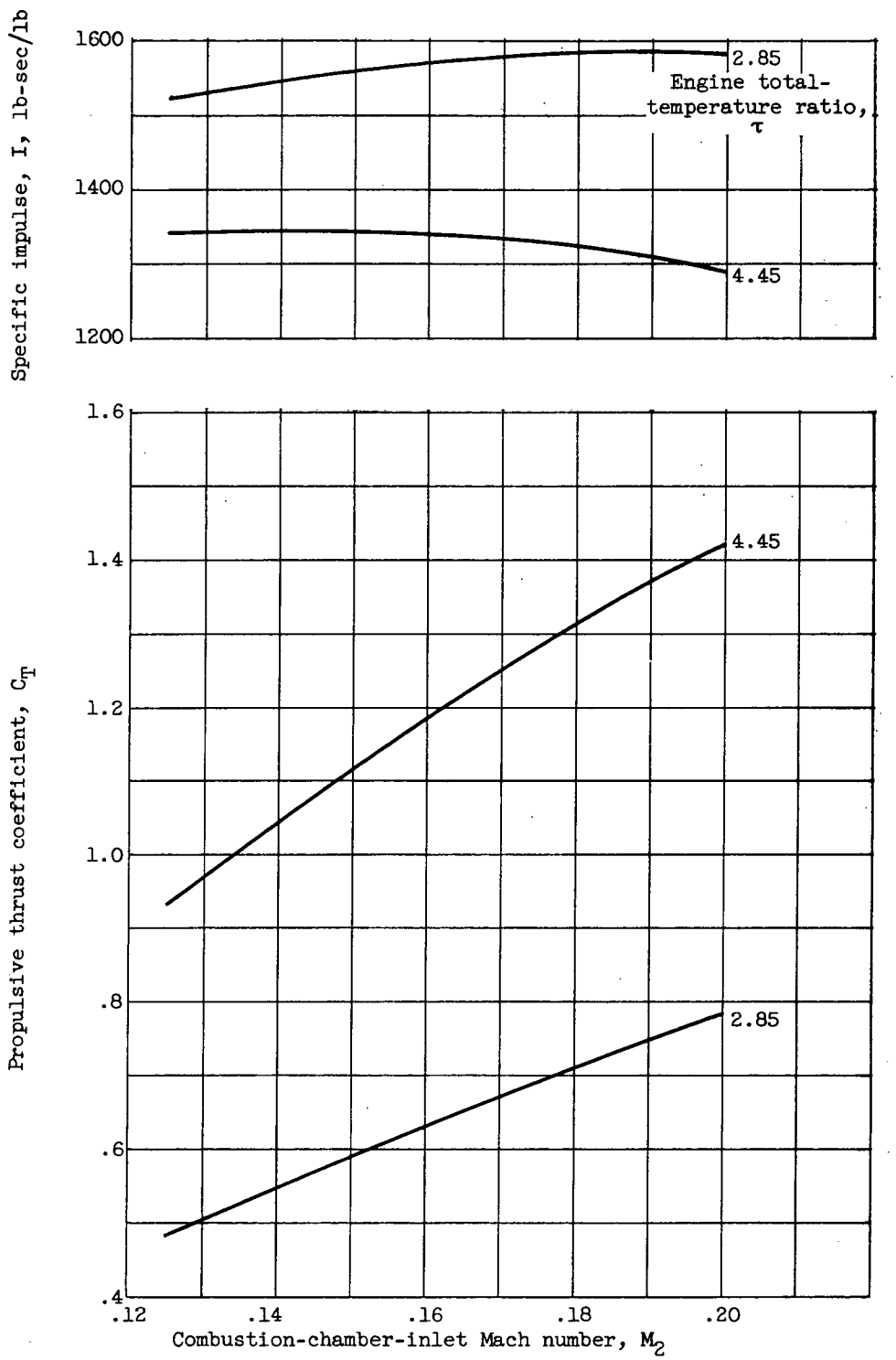



Figure 5. - Effect of flight Mach number on design-point performance of ram-jet engine. Engine total-temperature ratio chosen for maximum efficiency.

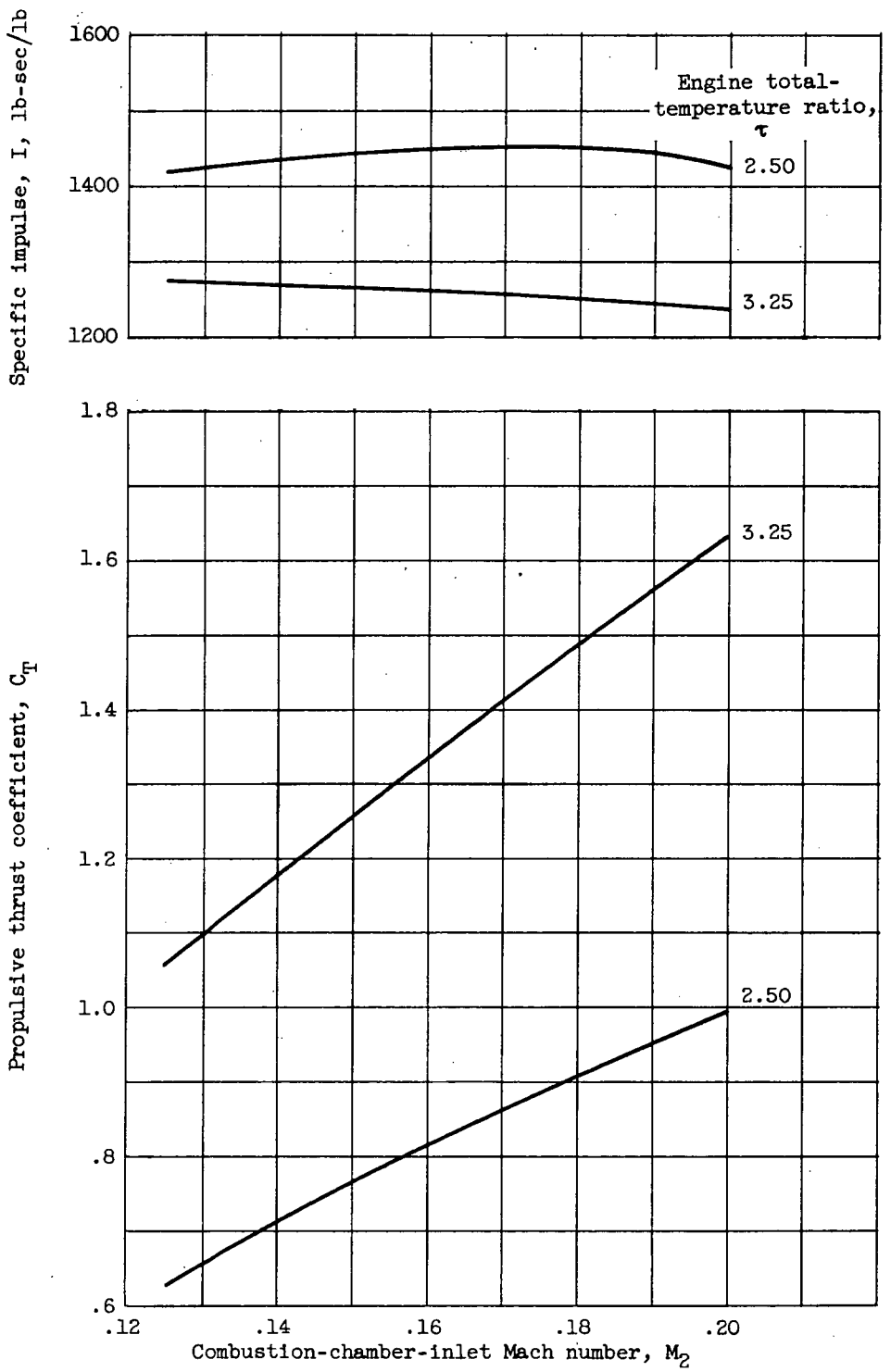
[redacted]




(a) Flight Mach number, 2.5.

Figure 6. - Effect of combustion-chamber-inlet Mach number on ram-jet engine performance.

[redacted]




(b) Flight Mach number, 3.5.

Figure 6. - Concluded. Effect of combustion-chamber-inlet Mach number on ram-jet engine performance.

[redacted]

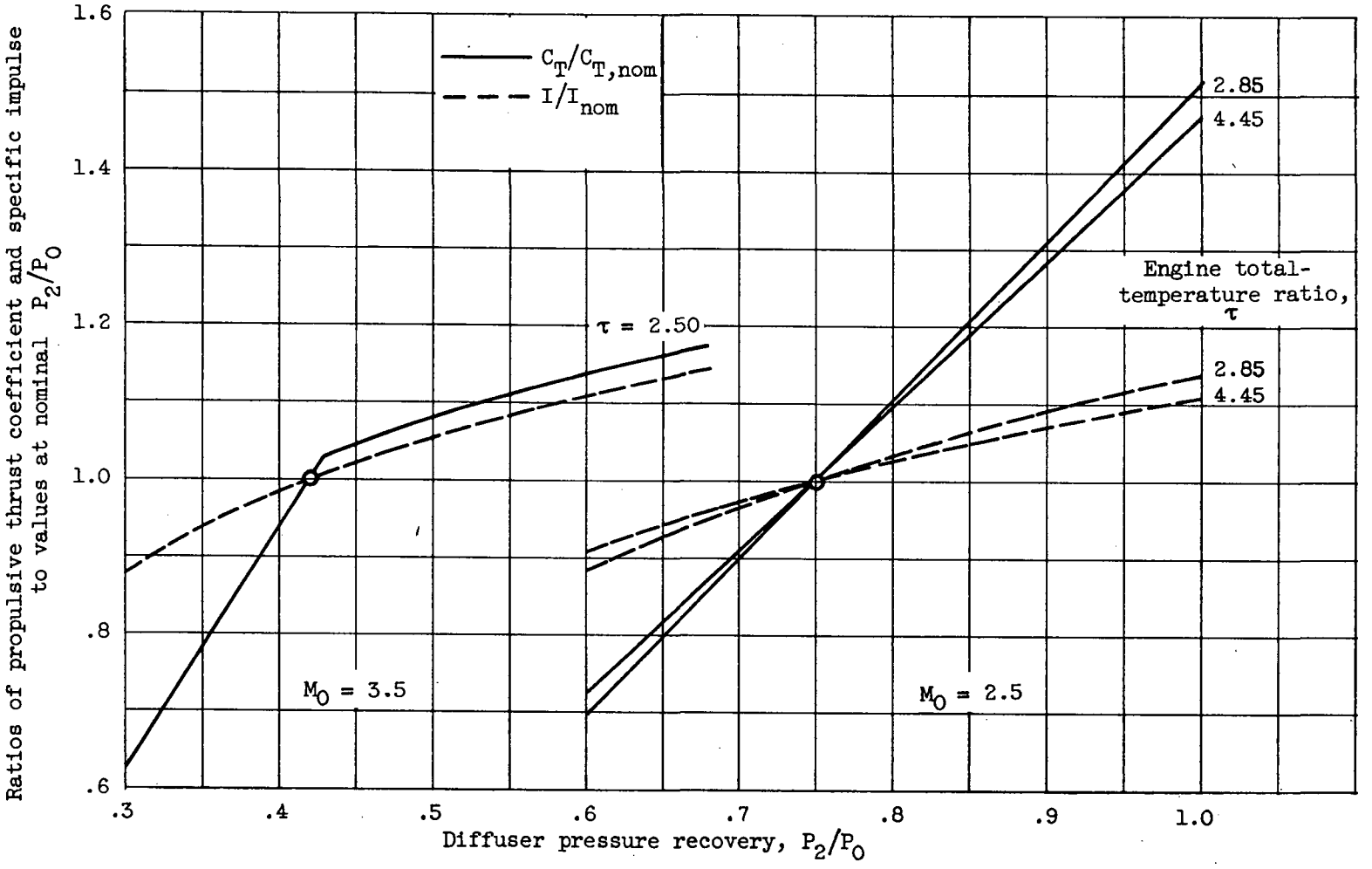



Figure 7. - Effect of diffuser pressure recovery on ram-jet engine performance. Combustion-chamber-inlet Mach number, 0.200.

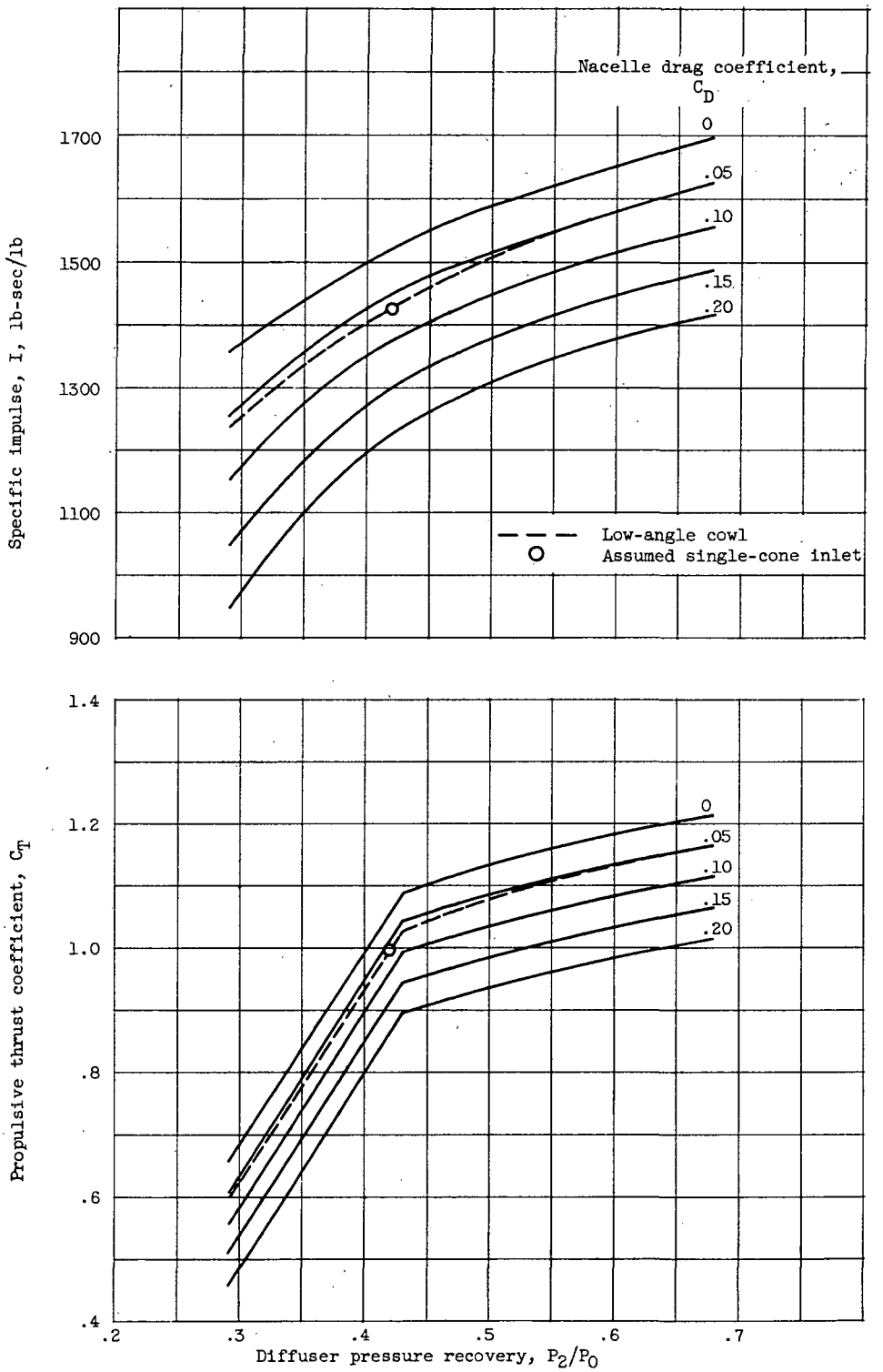



Figure 8. - Effect of diffuser pressure recovery and engine drag coefficient on ram-jet engine performance. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200; engine total-temperature ratio, 2.50.

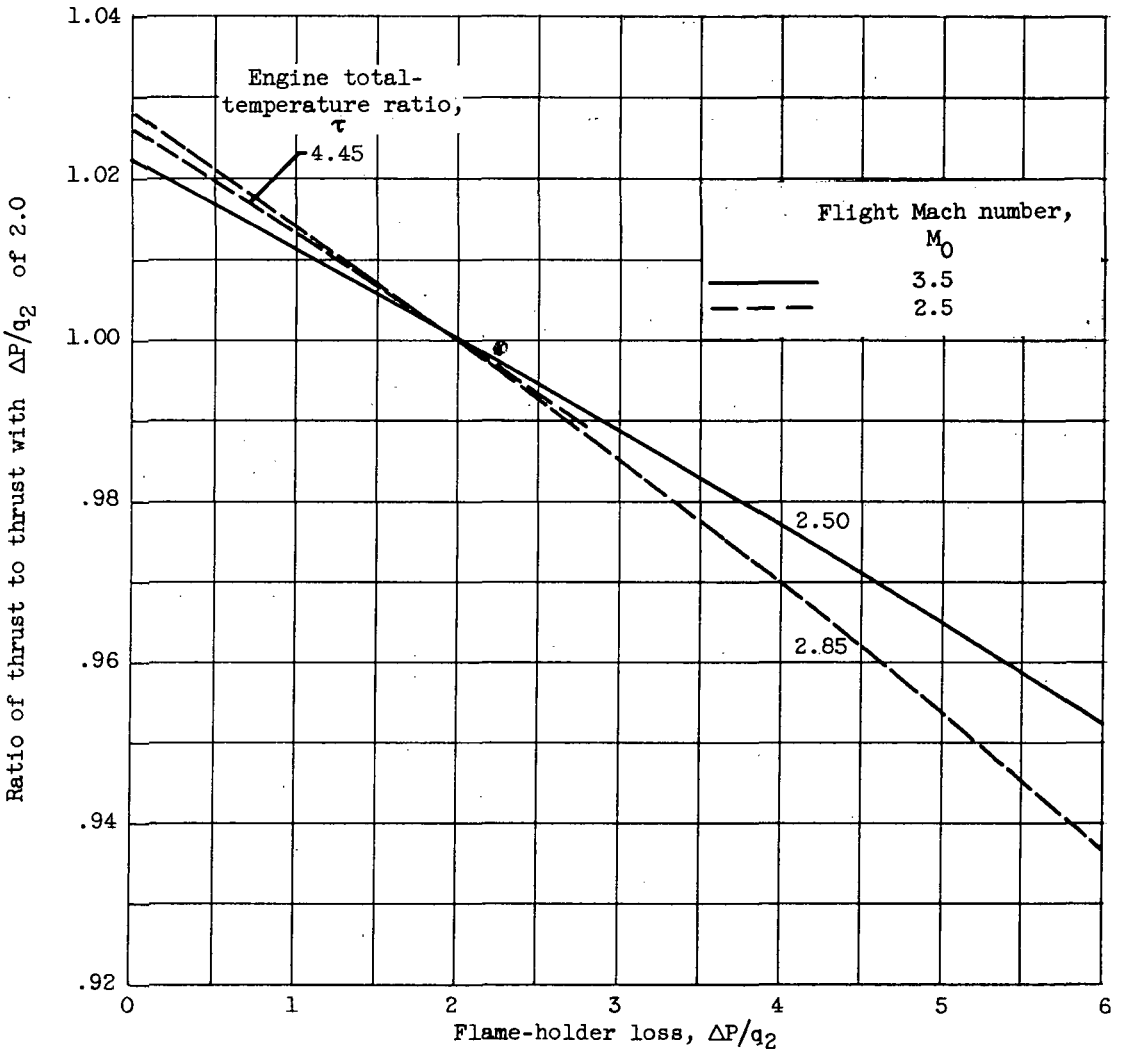



Figure 9. - Effect of flame-holder pressure loss on ram-jet engine performance. Combustion-chamber-inlet Mach number, 0.200; combustion efficiency, 0.90. (Specific impulse is affected in same proportion as thrust.)

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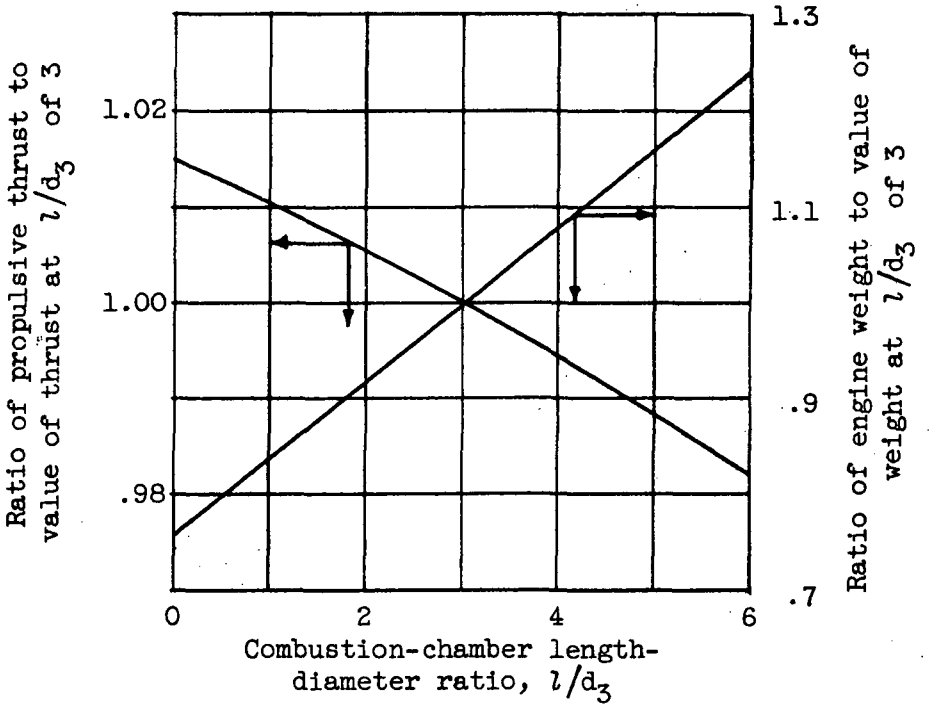


Figure 10. - Effect of combustion-chamber length on performance and weight of ram-jet engine. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200; engine total-temperature ratio, 2.50; combustion efficiency, 0.90. (Specific impulse affected in same proportion as thrust.)

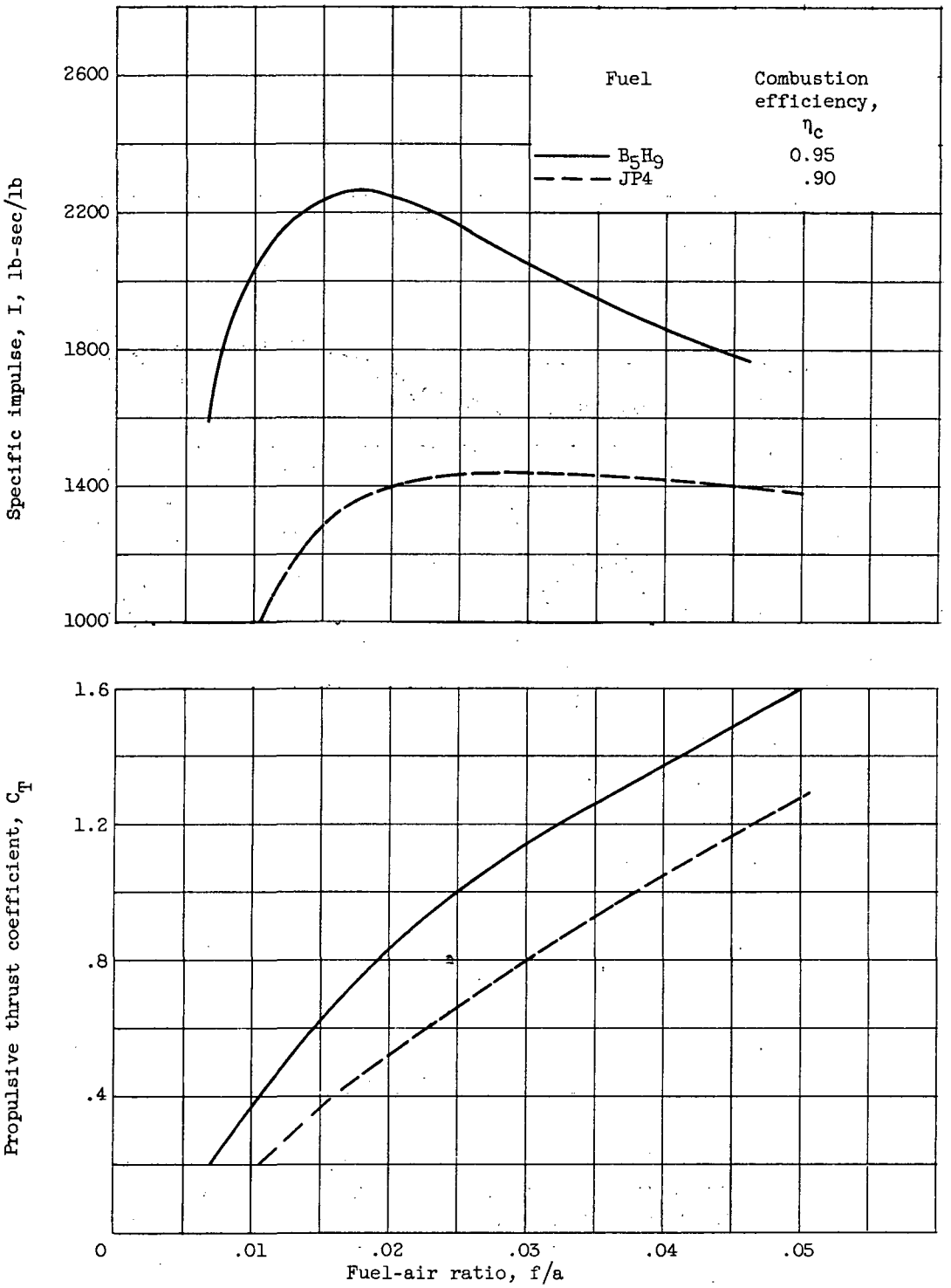


Figure 11. - Comparison between performance of engines using pentaborane and JP4 fuel. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200.

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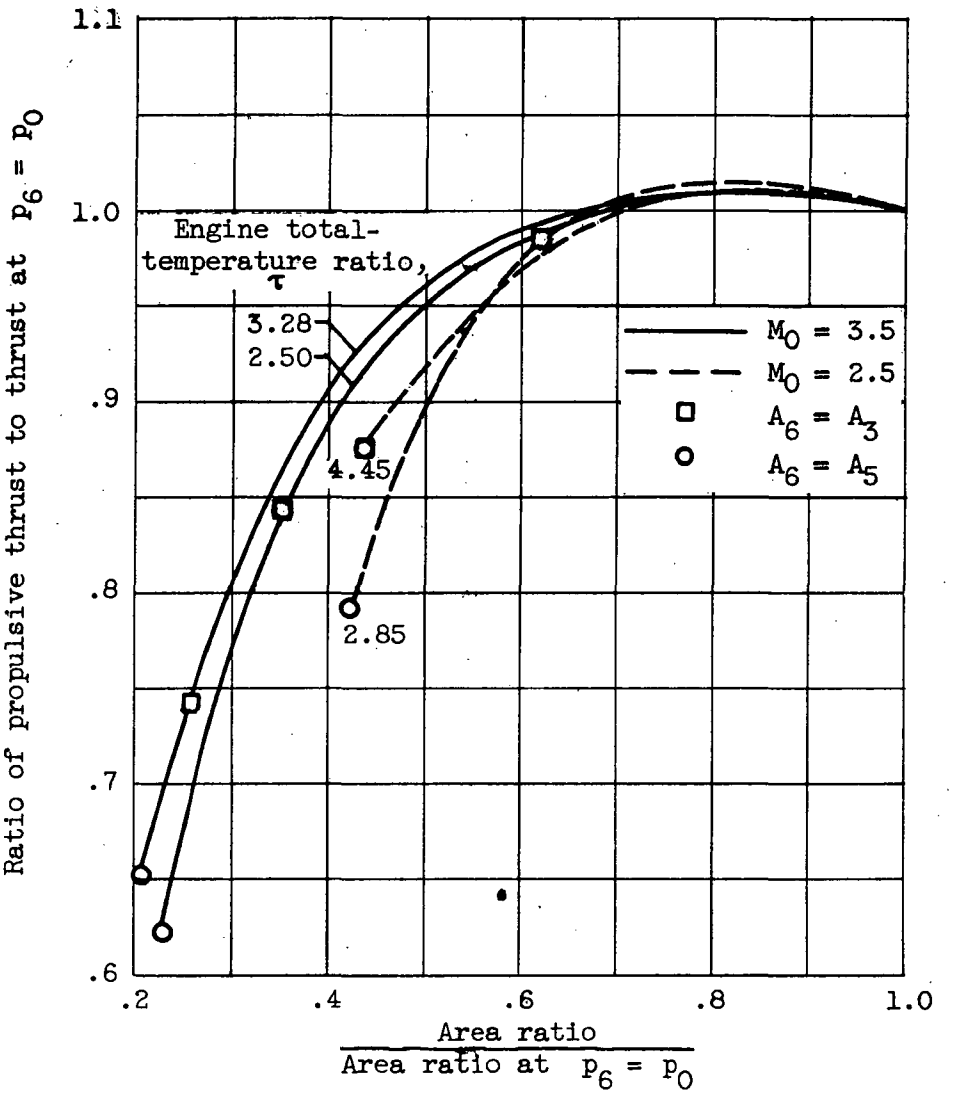


Figure 12. - Effect of nozzle area ratio on ramjet engine performance. Combustion-chamber-inlet Mach number, 0.200. (Specific impulse is affected in same proportion as thrust.)

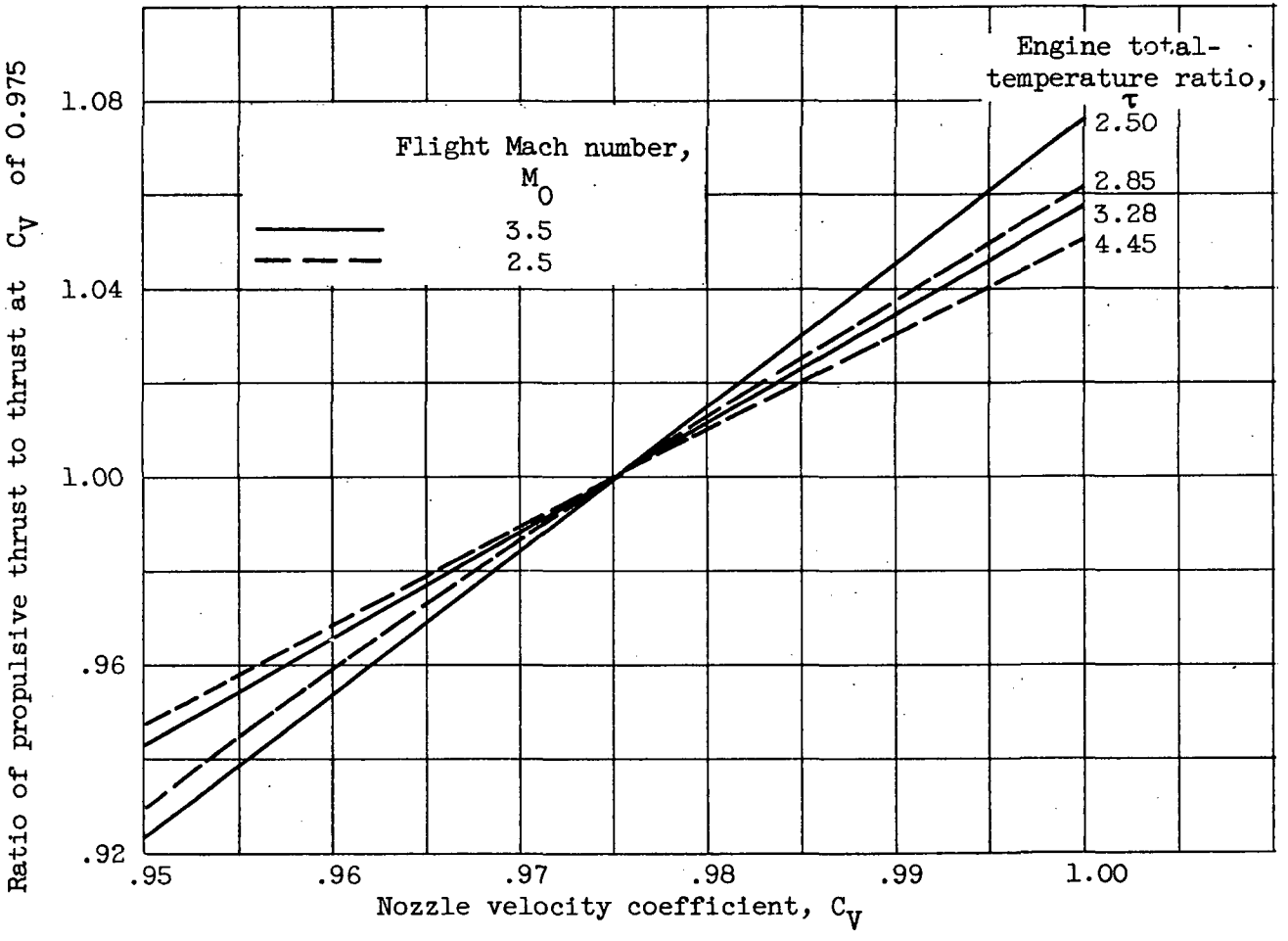


Figure 13. - Effect of nozzle velocity coefficient on ram-jet engine performance. Combustion-chamber-inlet Mach number, 0.200. (Specific impulse is affected in same proportion as thrust.)

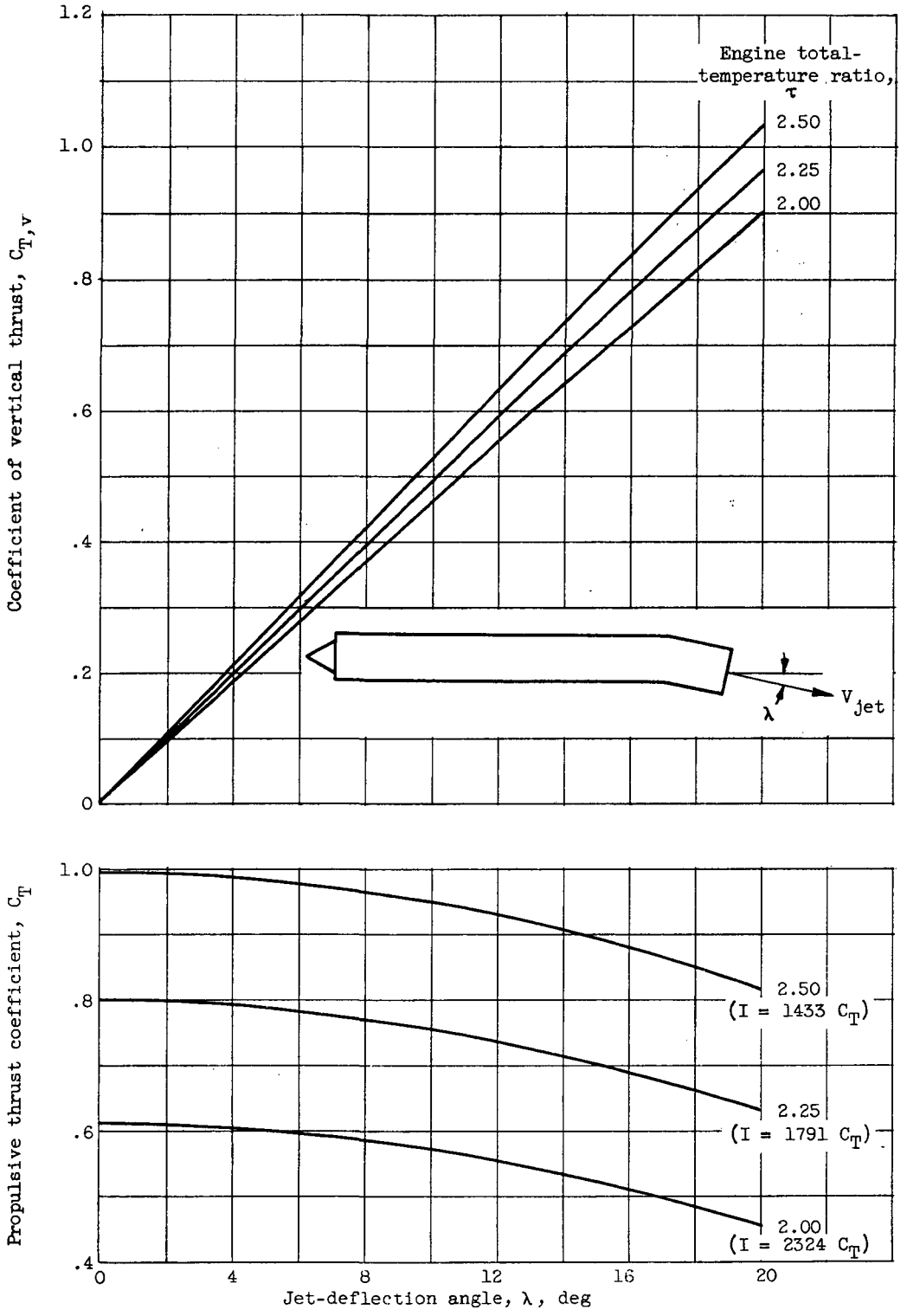


Figure 14. - Effect of jet-deflection angle on ram-jet engine performance. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200.

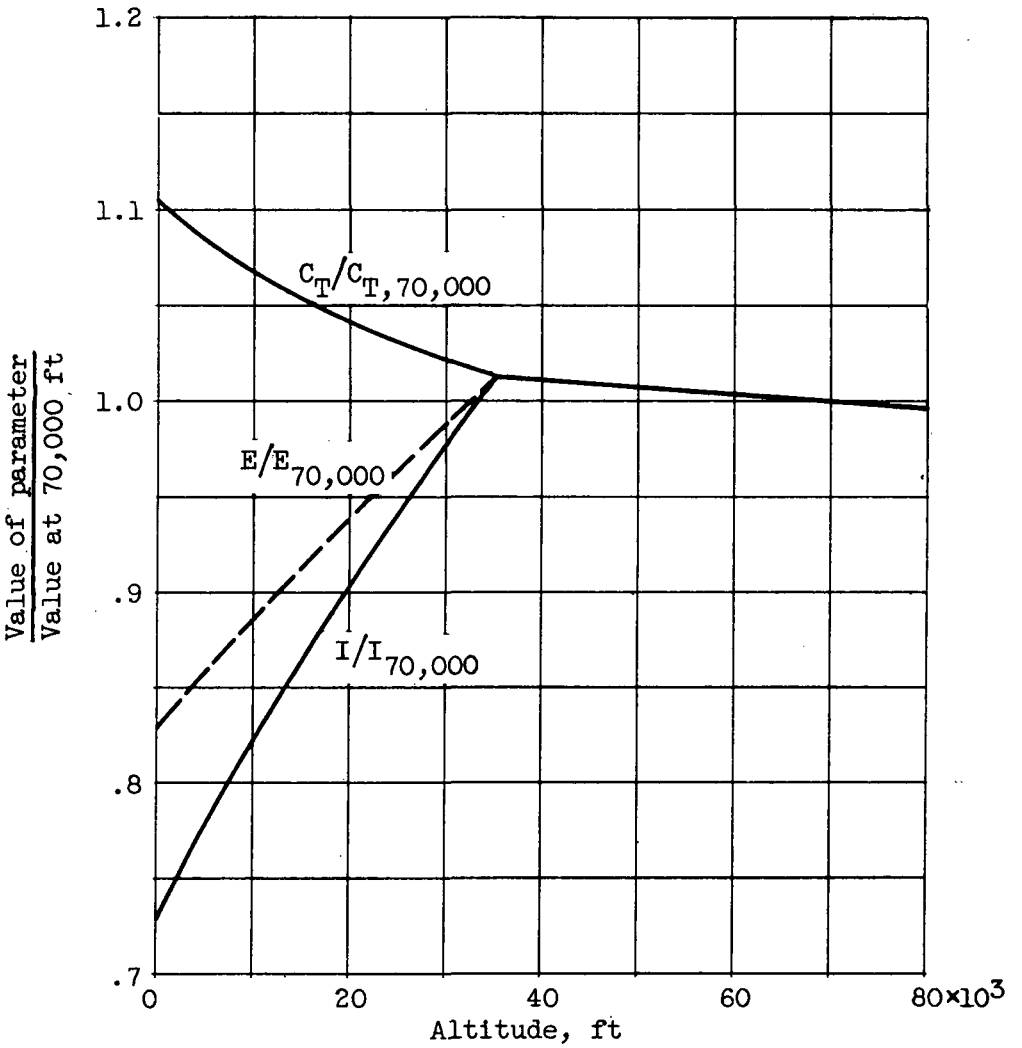


Figure 15. - Effect of altitude on ram-jet engine performance. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200; engine total-temperature ratio, 2.50.

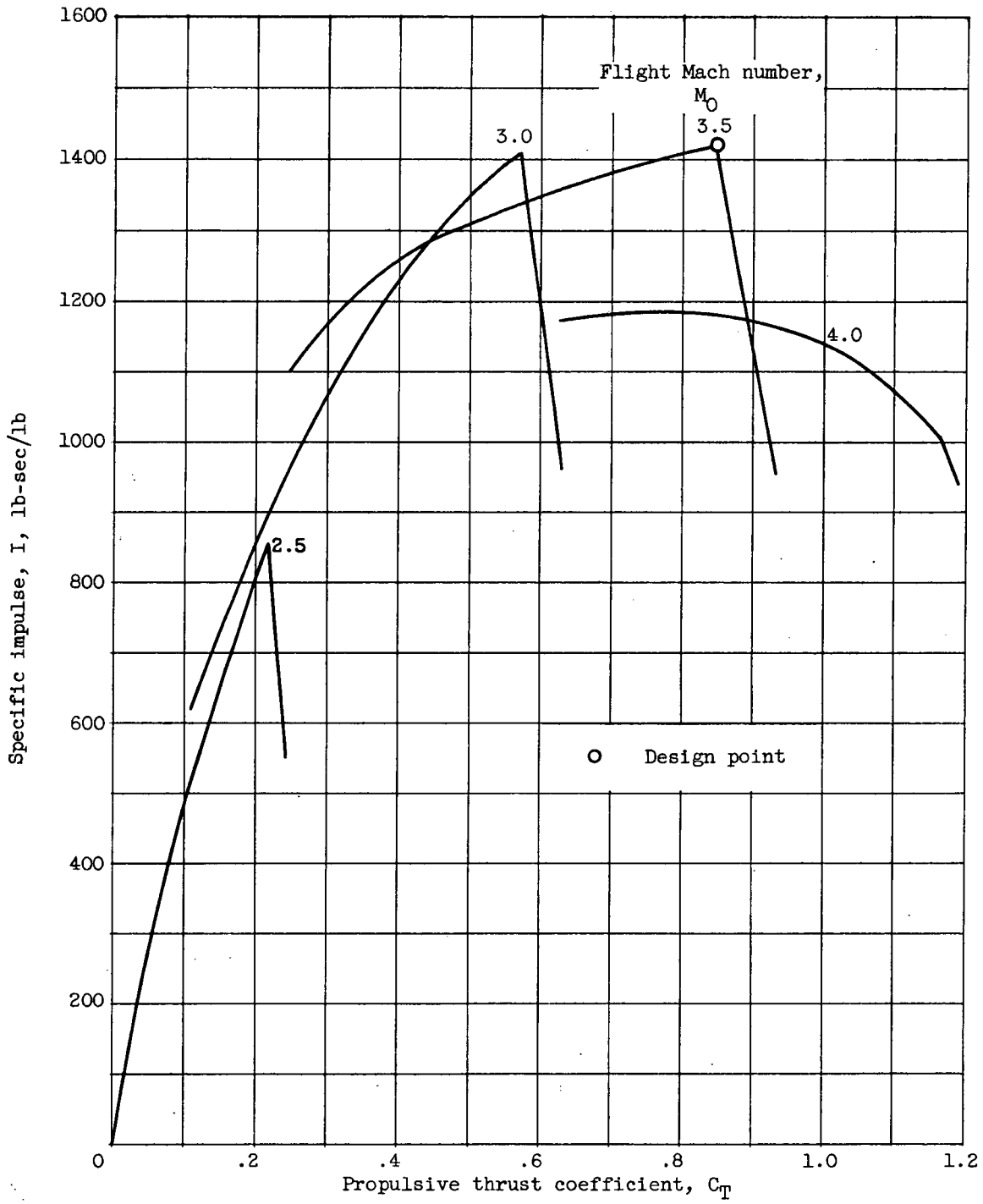


Figure 16. - Off-design performance of fixed-geometry ram-jet engine designed for flight Mach number of 3.5, combustion-chamber-inlet Mach number of 0.200, and engine total-temperature ratio of 2.25. Combustion efficiency, 0.87; ratio of specific heats for exhaust gases, 1.30.

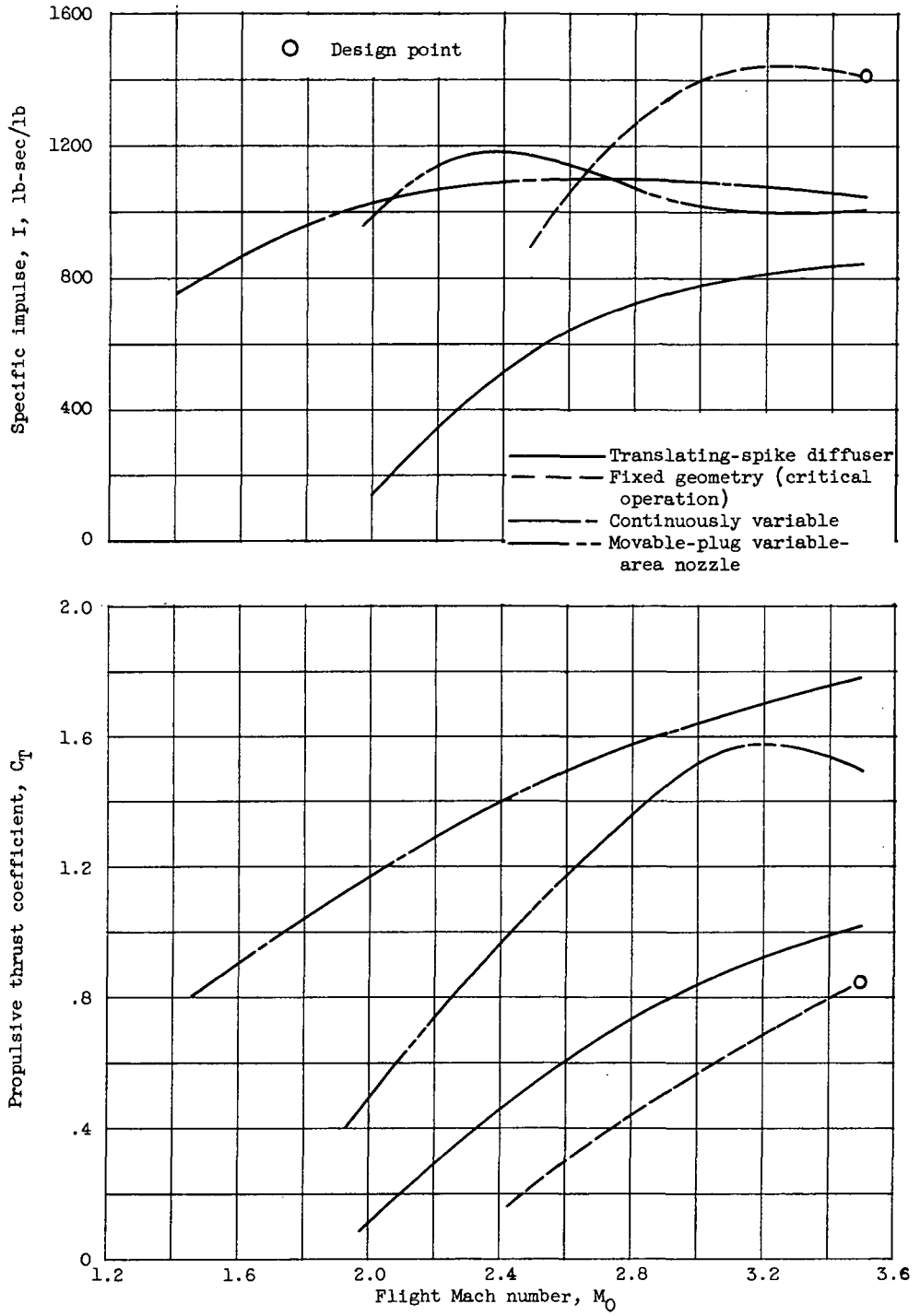


Figure 17. - Off-design performance of various fixed- and variable-geometry ram-jet engines operating at maximum thrust. Engines designed for flight Mach number of 3.5, combustion-chamber-inlet Mach number of 0.200, and engine total-temperature ratio of 2.50. Combustion efficiency, 0.87; ratio of specific heats for exhaust gases, 1.30.

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Unclassified when detached from rest of report

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